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ENERGY EFFICIENT ENGINE PROGRAM

TECHNOLOGY BENEFIT/COST STUDY
VOLUME II

by

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UNITED TECHNOLOGIES CORPORATION
Pratt & Whitney
Engineering Division

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FOREWORD

The Energy Efficient Engine Component Development and Integration program is being conducted under parallel National Aeronautics and Space Administration contracts with Pratt & Whitney and General Electric Company. The overall project is under the direction of Mr. Carl C. Ciepluch serving as NASA's project manager for the Pratt & Whitney effort under contract NAS3-20646. Mr. Frank Berkopec is the NASA project engineer responsible for the portion of the project described in this report. Mr. William B. Gardner is manager of the Energy Efficient Engine program at Pratt & Whitney.

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SECTION 1.0
SUMMARY

The National Aeronautics and Space Administration, under the Energy Efficient Engine Component Development and Integration program, sponsored the technology benefit/cost study with the objectives of:

1. identifying turbofan engine technology requirements for the years 2000 to 2010,

2. formulating programs for developing the technologies required for that time period.

This program identified a number of very attractive technology concepts that could yield thrust specific fuel consumption benefits of almost 16 percent relative to the Maximum Efficiency Energy Efficient Engine. These thrust specific fuel consumption advantages translate into fuel burn benefits of up to 24 percent and DOC+I benefits of up to 14 percent in a quadjet airplane. These concepts include:

- an advanced channel diffuser and combustor,
- advanced diffuser/combustor materials,
- a high efficiency high pressure turbine,
- high efficiency compressors,
- an advanced active clearance control,
- a high efficiency low pressure turbine,
- swept fan blades,
- a geared low pressure spool,
- an advanced nacelle.

The program consisted of six phases. The initial effort was to screen and rank preliminary technology concepts that might be amenable to future development. Cycle studies, flowpath definition studies, and mechanical feasibility studies were then used to establish the feasibility of critical elements of the technologies identified for 2000 to 2010 time frame. These efforts showed that a turbofan engine with advancements in aerodynamics, mechanical arrangements, and materials offered significant performance improvements over 1988 technology.

The fifth phase assessed the benefits of the technological concepts identified in the earlier phases using fuel burn and direct operating cost plus interest (DOC+I).

To realize the potential benefits of these technologies, detailed development programs that include the scope of work, technical approach, schedule and cost for completion have been recommended to the government.

SECTION 2.0
INTRODUCTION

2.1 BACKGROUND

The National Aeronautics and Space Administration has the objective of improving the energy efficiency of future United States commercial aircraft so that substantial savings in fuel can be realized. To achieve this objective, NASA established the Energy Efficient Engine Component Development and Integration program in 1978 under contract NAS3-20646. Minimum goals for this program are a 12 percent reduction in thrust specific fuel consumption (TSFC) and a 5 percent reduction in direct operating costs (DOC) compared to the Pratt & Whitney JT9D-7A engine. In addition, FAR Part 36 (1978) noise rules and EPA-proposed 1981 exhaust emissions standards must be met.

The Energy Efficient Engine Component Development and Integration program is based on the results of the completed Energy Efficient Engine Preliminary Design and Integration study, NASA Contract NAS3-20628, described in NASA CR-135396. Through the extension of the technology base developed under the earlier program, the Energy Efficient Engine Component Development and Integration program will develop and demonstrate the technology for achieving higher thermodynamic and propulsive efficiencies in future environmentally acceptable turbofan engines.

To meet these program objectives, the current program consists of the following two tasks.

Task 1 - Flight Propulsion System Analysis, Design, and Integration

Task 2 - Component Analysis, Design and Development

More specifically, Task 1 consists of six subtasks:

- propulsion system preliminary design,
- control preliminary definition,
- propulsion system analysis and design update,
- propulsion system/aircraft integration evaluation,
- program risk assessment, and
- technology benefit/cost study.

The sixth subtask, the technology benefit/cost study, was a 1981 addition to the program to consider advanced turbofan technologies beyond the current Energy Efficient Engine System. This report presents the results of the technology benefit/cost study. Volume I of this report is an Executive Summary of the entire subtask published under separate cover. This volume, Volume II, describes the efforts conducted under the subtask and presents the conclusions of this subtask. A separate, detailed compendium of the key technology development plans formulated in the last phase of the subtask has been provided to the government.

2.2 SCOPE OF EFFORT

Development of technology for gas turbine engine propulsion systems from concept formulation to full scale demonstration is the combined result of Government sponsored research, exploratory development, advanced development programs; and corporate research and development programs. These various sources provide an expanded, improved technology base which can be applied to a broad spectrum of advanced systems when required.

Due to the long lead time for technology development, early projections of future propulsion system technology requirements are necessary to ensure technological maturity when advanced commercial and military aircraft are needed. Experience indicates that it takes about four years to identify a new idea, relate it to a future program, and obtain the necessary support for demonstrating its feasibility. It then requires approximately four more years to demonstrate an advanced concept and develop the design tools required to apply the concept to an engine design with reasonable confidence. Therefore, this technology benefit/cost study effort was initiated with the objectives of:

1. identifying turbofan engine technology requirements for the years 2000 to 2010,
2. formulating programs for developing the technologies required for that time period.

The results of this study verified that there are still large potential benefits to be realized from the advancement of gas turbine engine technology. While the primary interest of the Energy Efficient Engine Component Development and Integration program is improved fuel efficiency for commercial aircraft engines, the technology envisioned as a result of this study may also be applicable to military engines. For example, a primary performance consideration with fighter engines is the thrust to weight ratio. Materials advancements with lighter, stronger materials will not only lead to higher thrust to weight ratios, but can also lead to greater range, payload, and fuel efficiency in both commercial and military applications. The same benefits could be achieved from advancements in rotor speeds which would enable the reduction of airfoil count and, therefore, weight while maintaining the same thrust and efficiency.

2.3 STUDY APPROACH

To meet the study objectives and identify the technologies that could potentially provide increased fuel efficiency and other benefits, the Benefit/Cost Study subtask was structured into six phases:

- screen preliminary technologies,
- perform cycle studies,
- define flowpaths of candidate engines,
- establish mechanical feasibility of key technological concepts,
- perform benefit/cost analysis,
- establish key technology development plans.

A description of each study phase is presented below.

2.3.1 Screen Preliminary Technologies

The intent of this phase was to screen and rank preliminary technology candidates based on their potential fuel savings and potential operating cost reduction compared to a reference engine. The screening started with identification of the functional benefits; i. e., performance, weight, cost, environmental performance, durability, maintainability, and reliability. This process also considered the means of achieving the benefit provided by the concept; i. e., direct substitution, cycle changes and or configurational modification to an existing engine; and the technology development requirements. These candidates would be used as guidance for further refinement of technology projections and determination of final technology concepts in the remaining phases of this subtask. Section 3.0 presents the results of this phase of the subtask.

2.3.2 Perform Cycle Studies

The initial effort was to review the historical trends of gas turbine engine pressure ratio, combustor exit temperature, and overall efficiency. Projections were made to define the expected levels of component and subsystem efficiency in the 2000 to 2010 time period. Cycle studies were then conducted in which overall pressure ratio, combustor exit temperature, fan pressure ratio and bypass ratio were varied. These studies resulted in identification of eight possible engine cycle candidates and the selection of nine final technology concepts for further evaluation in the flowpath definition and mechanical feasibility studies. Section 4.1 presents the results of the cycle studies.

2.3.3 Define Flowpaths of Candidate Engines

The flowpath studies produced gaspath geometry to be used in the subsequent mechanical feasibility evaluation of critical technological elements. These studies iterated such characteristics as rotor speeds, airfoil geometry, inlet and exit diameters, and spool splits until the levels of component and subsystem efficiency projected in the cycle studies were achieved. From the eight candidates defined in the cycle studies, these efforts identified three flowpaths that had the potential to meet projected efficiency requirements. A description of the flowpath definition studies is presented in Section 4.2.

2.3.4 Establish Mechanical Feasibility

The mechanical feasibility studies evaluated some of the critical elements of the technologies needed to meet expected efficiency levels of the 2000 to 2010 time period. Critical areas of each of the three final flowpath candidates, such as number of spools, clearance control, nacelle configurations, rotor support arrangements and mounting, were analyzed. Section 4.3 presents the results of the mechanical feasibility studies.

2.3.5 Perform Benefit/Cost Analyses

The benefit/cost analyses determined the benefits expected from the advanced technologies identified in the cycle studies, flowpath studies and mechanical feasibility studies. These technologies were compared to a reference engine for projected benefits in mission fuel burn and direct operating cost plus interest (DOC+I). Three airplane types were configured with both the reference engine and the advanced configuration for comparison and three levels of fuel price were assumed. The results of the benefit/cost analyses are presented in Section 5.0.

2.3.6 Establish Key Technology Development Plans

To realize the potential benefits identified in the benefit/cost analysis, the technologies required for the 2000 to 2010 time period engines must be developed. In this phase of the subtask, detailed plans were established for the development of the both the major and supporting technologies. This effort is summarized in Section 6.0. Detailed technology development plans including objective, scope of work, schedule, approach, and estimated cost to complete have been provided to the government.

SECTION 3.0
IDENTIFICATION, SCREENING AND ANALYSIS OF PRELIMINARY TECHNOLOGY CANDIDATES

3.1 IDENTIFICATION AND SCREENING OF PRELIMINARY TECHNOLOGY CANDIDATES

To provide direction for the establishment of the technological concepts needed to meet the engine requirements of the 2000 to 2010 time period, an initial identification and screening of preliminary technology candidates was conducted. A series of candidates was selected by a method typically used for technology planning. In this procedure, a preliminary listing was prepared of candidates that had the potential to provide fuel consumption benefits in future engines.

In a screening process, these preliminary candidates were reviewed, discussed and revised by experts in each appropriate technical discipline. Screening of these preliminary candidates included the determination of the functional benefits of each candidate; i. e., performance, weight, cost, environmental performance, durability, maintainability, and reliability. The screening procedure also considered the means of achieving the benefit provided by the candidate, i. e., direct substitution, cycle changes and/or configurational modification to a reference engine; and the technology development requirements.

The relevance of these candidates to future product requirements was then assessed. This procedure resulted in a list of preliminary technology candidates selected for the benefit analysis described below.

3.2 BENEFITS OF PRELIMINARY TECHNOLOGY CANDIDATES

The preliminary technology candidates were analyzed for their potential fuel savings and potential direct operating cost reductions when incorporated into the benefit/cost study reference engine. Section 3.2.1 presents a description of the benefit/cost study reference engine, against which potential benefits were measured. Section 3.2.2 defines direct operating cost as used in the analysis of these preliminary candidates and Section 3.3.3 presents the results of the analyses.

3.2.1 Benefit/Cost Study Reference Engine

Benefits of the preliminary advanced technology candidates were quantified by comparing the performance of the benefit/cost study reference engine with and without incorporation of those technology candidates. The benefit/cost study reference engine is the Maximum Efficiency Energy Efficient Engine configured in 1981 under other efforts in the Energy Efficient Engine Component Development and Integration program (Reference 2.0). This engine configuration represents a reoptimization of Energy Efficient Engine technology to reflect major increases in fuel price between 1978 and 1981. It includes a high bypass ratio and features high efficiency components designed to substantially improve fuel economy and direct operating costs over the flight propulsion system developed earlier in the Energy Efficient Engine Component Development and Integration program. With fuel costs dominating current airline operating economics, the reference engine provided an estimated improvement of five percent in cruise thrust specific fuel consumption compared to the flight propulsion system.

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Compared to the flight propulsion system, the reference engine includes a higher bypass ratio single stage fan with a larger diameter 7.6 cm (3.0 in), a ten stage high pressure compressor with reduced axial gapping, a simpler one stage combustor to meet proposed or projected emissions requirements, and two additional turbine stages for a substantial improvement in component efficiencies. A comparison of the flight propulsion system and the reference engine is presented in Figure 3.2-1. Figure 3.2-2 shows the reference engine installed in a mixed exhaust nacelle system. The reference engine is mounted in the same manner as the flight propulsion system with front thrust links situated at the engine centerline horizontally and the rear mount system located at the front of the low pressure turbine. Flight loads are shared between the engine and nacelle structure.

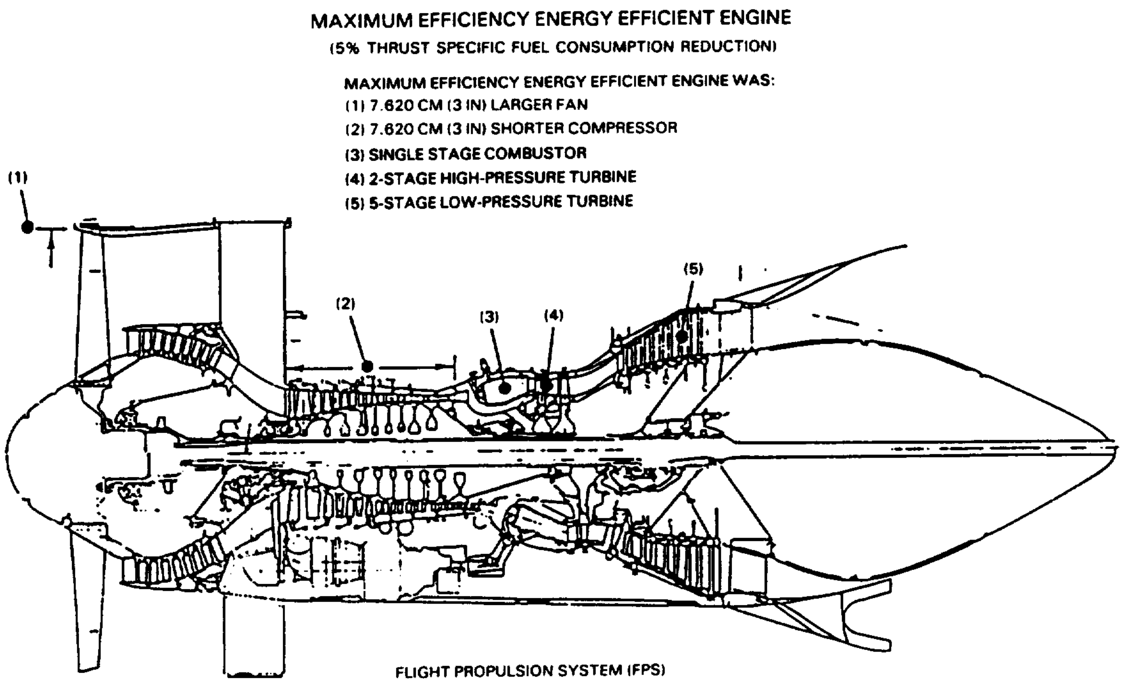

Figure 3.2-1 Reference Engine Differences from the Flight Propulsion System

3.2.1.1 Overall Cycle

The overall cycles of the two configurations are compared in Table 3.2-I. Engine overall pressure ratio and turbine inlet temperature levels of the reference engine were not changed from the flight propulsion system to be consistent with Energy Efficient Engine program materials and cooling technology. However, bypass ratio was re-examined due to higher fuel prices than those used when the flight propulsion system was configured in 1977. Fuel prices ranged from 40 to 45 cents per 3.78 liters (1.0 U.S. gallon) in 1977, while a more representative level of $1.50/3.78 liters was used in the 1981 reference engine development. Because of this difference in fuel costs, the bypass ratio needed to minimize direct operating costs increased from 6.5 to 7.2.

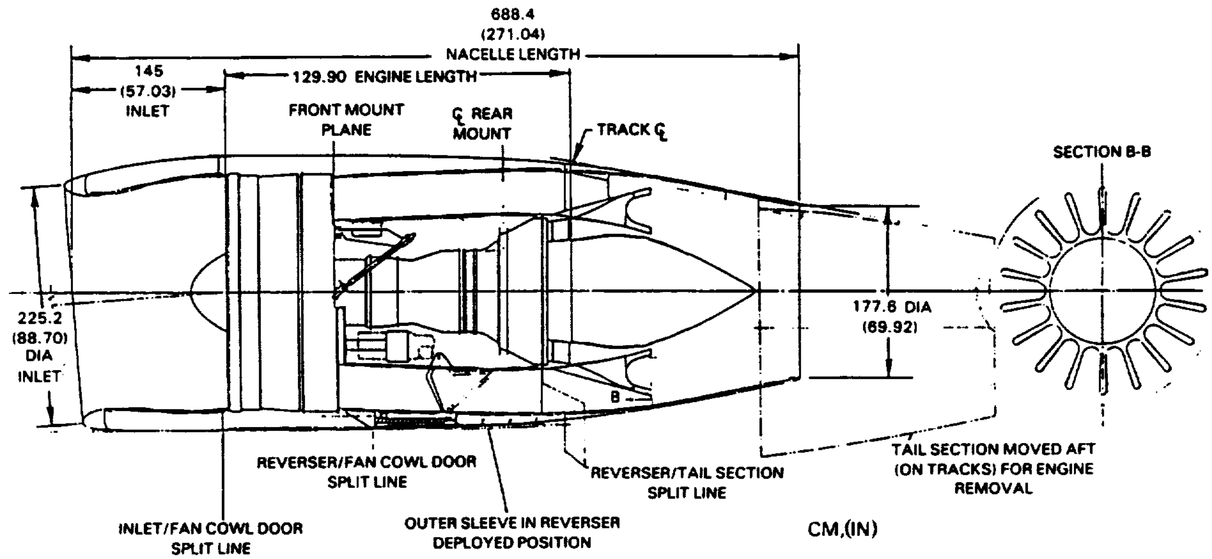

Figure 3.2-2 Reference Engine In Advanced Nacelle System

TABLE 3.2-I
COMPARISON OF REFERENCE ENGINE OVERALL CYCLE TO
FLIGHT PROPULSION SYSTEM CYCLE
(10,675 m (35,000 ft), 0.8 Mn, Max Cruise)

	Flight Propulsion System (1977)	Reference Engine (1981)
Bypass Ratio	6.5	7.2
Overall Pressure Ratio	38.6	38.6
Combustor Exit Temperature °C (°F)	1268 (2314)	1268 (2314)

3.2.1.2 Component Aerodynamic Design Differences

The flight propulsion system components were modified to accommodate the higher bypass ratio of the reference engine. The duct exit guide vane (DEGV) area ratio, i. e., the inlet area divided by the exit area, was increased by two percent on the higher bypass ratio fan to reduce the inlet Mach number and the aerodynamic loadings on the vanes.

Fan inner diameter pressure ratio was set by holding the root work coefficient of the flight propulsion system. Since the fan is slowed relative to the base, the inner diameter pressure ratio is lower than the flight propulsion system, resulting in a higher low pressure compressor pressure ratio.

The higher pressure ratio in the low pressure compressor required the exit Mach number to be raised in conjunction with exit elevation to hold the same surge margin as the flight propulsion system.The intermediate case length of the flight propulsion system was set structurally (inlet guide vane chord, strut chord, axial gapping, etc.) resulting in an aerodynamically unloaded design. However, the bearing compartment in the reference engine was redesigned by placing the centering spring directly under the bearing. This resulted in an inlet guide vane chord reduction and tighter gapping between the strut and high pressure compressor rotor leading edge, as shown in Figure 3.2-3.

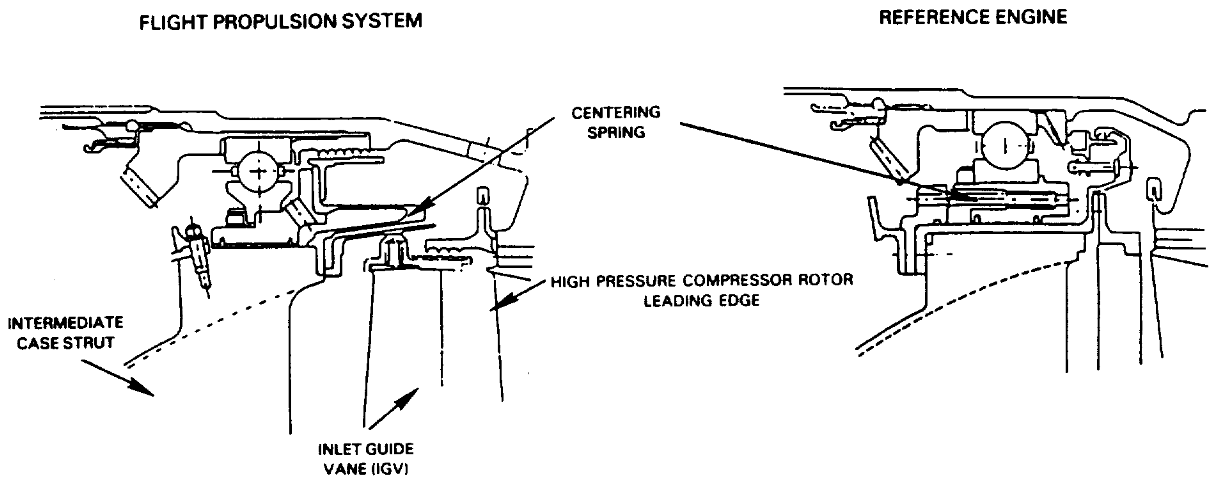

Figure 3.2-3 Centering Spring Rearranged to Shorten Intermediate Case in Reference Engine

The high pressure compressor of the reference engine remained aerodynamically unchanged from the flight propulsion system. The only difference is a length reduction of 7.3 cm (2.9 in) as shown in Table 3.2-II.

TABLE 3.2-II
ALTERATIONS TO FLIGHT PROPULSION SYSTEM FOR SHORTER HIGH PRESSURE COMPRESSOR IN REFERENCE ENGINE

Removal of flowguides - cm (in)	-1.77 (-0.70)
Incorporation of shorter bleed ports - cm (in)	-0.50 (-0.20)
Elimination of variable vane provisions in stages 9-15 (experimental requirement)-cm(in)	-1.52 (-0.60)
Elimination of excessive gaps - cm (in)	-3.55 (-1.40)
Total Reduction	-7.36 (-2.90)

The combustor in the reference engine was changed from the two stage design of the flight propulsion system to a single stage aerating design. This change was consistent with regulations on future emissions levels projected by both the International Civil Aviation Organization and the Environmental Protection Agency.

The reference engine combustor has a 1.1 percent lower pressure loss than the flight propulsion system due primarily to the use of a straight wall diffuser rather than a curved-wall diffuser. Also, a 3.3 cm (1.3 in) length reduction resulted from the use of a smaller diameter two stage high pressure turbine which reduced the combustor cant angle.

A two stage high pressure turbine in the reference engine replaced the single stage design in the flight propulsion system. The combination of an added high pressure turbine stage and higher bypass ratio resulted in a larger radial offset of the turbines in the reference engine. Therefore, the transition duct cant angle was increased to 25 degrees and resulted in a transition duct 3.6 cm (1.4 in) longer than that of the flight propulsion system.

The five stage low pressure turbine in the reference engine was configured with the same level of blade turning, exit axial Mach number, and maximum diameter as the flight propulsion system four stage design. With the added stage, the mean velocity ratio was increased and the exit swirl angle was reduced for increased efficiency.

3.2.1.3 Benefits of Reference Engine

The variations of the reference engine from the flight propulsion system provided an estimated 5 percent improvement in thrust specific fuel consumption. This improvement translates into a substantial savings in direct operating costs comparisons since fuel in 1981 constituted over 50 percent of the total direct operating cost.

3.2.2 Direct Operating Cost

A primary component of direct operating cost as used in the initial screening and ranking was fuel cost. All fuel efficiency analyses on the preliminary technology candidates used Pratt & Whitney's fuel cost escalations to the year 2000. These escalations were based on a year-by-year projection of the Industrial Commodities Wholesale Price Index to the year 1990. From 1990 to the year 2000, a nine percent per year inflation rate was assumed. A three percent per year fuel price escalation was superimposed on these generalized rates.

Other components of direct operating cost include crew cost, utilization costs and airframe maintenance cost. Each of these three costs were calculated using a 1981 Boeing Commercial Airplane Company method. The cost of engine maintenance was calculated using a standard Pratt & Whitney calculation for a mature engine model.

The remaining components of direct operating cost include maintenance burden (200 percent on labor), airplane/engine price, insurance, spares, and depreciation.

3.2.3 Results of Savings Analysis

Forty-three candidates were selected and analyzed for their potential fuel savings and potential direct operating cost reductions when incorporated into the benefit/cost study reference engine. Results from these analyses, presented in Table 3.2-III, indicate the majority of candidates offer fuel burned and direct operating cost reductions of 0.1 to 0.5 percent. These relatively small differences make a justifiable definition of rankings impossible. Therefore, criteria used for selecting candidates for guidance in the cycle studies, flowpath definition and mechanical feasibility analyses were revised to include those candidates that:

1. offer fuel savings and direct operating cost reduction,

2. represent an evolutionary extension of current program technology rather than an innovative design approach, and

3. are amenable to development programs.

The result of this decision was that all of these preliminary technology candidates, with the exception of those that show no fuel burn benefit, were used as guidance in the establishment of a flight configuration for the 2000 to 2010 time period.

TABLE 3.2-III
PRELIMINARY TECHNOLOGY CANDIDATE SCREENING SUMMARY

Candidate	Percent TSFC Reduction	Percent Block Fuel Savings	Percent DOC Reduction
Fan			
Shroudless, Hollow fan Blade	0.6	0.6	0
Tuned Fan Blade	0.6	0.3	0.2
Reduced Hub/Tip Ratio	0.4	0.5	0.3
Swept Fan Blade	0.5	0.5	0.1
Fan Blade Clearance Adjustment	0.1	0.1	0.1
Fan Exit Guide Vane Endwall Suction	0.1	0.2	0.1
Compressor			
Radial Work Endwall Improvement	0.2	0.2	0.1
Second Generation Controlled Diffusion Airfoils	0.2	0.2	0.2
Pressurized Inner Seal Cavities (Dropped from Study)			
Variable Compression (Alternative Cycle Change Simpler)			
Centrifugal Compressor	0		
Integrated Exit Guide Vane/Strut			
Combustor			
MARK IV Combustor	1.2	1.4	0.7
Advanced Segmented Liner	1.0	1.2	1.0
High Pressure Turbine			
Leaned/Bowed Vanes	0.5	0.6	0.4
Increased AN^2	0.4	0.4	0.2
Increased Efficiency Blade Cooling	0.1	0.1	0.1
Airfoil Thermal Barrier Coating	0.3	0.4	0.3
Single Crystal-1000 Vane with PS200 Coating	0.3	0.4	0.3
Single Crystal-2000 Vane with PS200 Coating	0.1	0.1	0.1
Single Crystal-3000 Vane with PS200 Coating	0.1	0.1	0.1
Multi-Piece Vane	0	0	0.2
Low Pressure Turbine/Exhaust Mixer			
Improved Mixer	0.2	0.2	0.1

TABLE 3.2-III (Continued)

Candidate	Percent TSFC Reduction	Percent Block Fuel Savings	Percent DOC Reduction
Air Management			
Modulated TOBI (tangential on board injection) System	0.1(Cruise)	0	0
Modulated Combustor Air	0.1(Cruise)	0	0
Modulated Vane Cooling Flow	0.4(Cruise)	0.2	0
Radial Flow TOBI	0	0	0
Optimized Customer Bleed	0.7	0.9	0.5
Closed Loop Active Clearance Control	0.5(Above 20,000 ft)	0.4	0.1
Precooled Turbine Cooling Air with Fuel Coolant	0.4	Deleted due to coking potential	
Improved Low Pressure Turbine Active Clearance Control	0.2	0.3	0.3
Installation			
Low Pressure Loss Duct	0.2	0.2	0.1
Low Isolated Drag Nacelle	0.7	0.9	0.6
Low Interference Drag Installation	3.0	3.6	2.2
Nacelle Vent Thrust Recovery	0.2	0.2	0.1
Engine Torque at Front Mount			
Variable Jet Area	0.1	0.2	Unknown
All Electric Power Extraction	1.5	1.8	1.1
Structures/Mechanics			
Composite Fan Cases	0	0.1	0.1
Composite Intermediate/Fan Exit Case	0	0.2	0.1
Integrated Fan Containment/Nacelle Inlet	0	0.1	0.05
Composite Core Cowl	0	0.1	0
High Efficiency Reduction Gear	3.0	3.1	1.6

SECTION 4.0
ENGINE CONFIGURATION AND TECHNOLOGY REQUIREMENT IDENTIFICATION

4.1 CYCLE STUDIES

An important phase in determining long range technology requirements was the selection of the major operating cycle for an engine configuration of the projected time period. In this phase of the subtask, several potential engine cycles and nine required advanced technologies were identified for further evaluation in the flowpath definition and mechanical feasibility studies. This selection process covered evaluation of the gas generator overall pressure ratio (OPR), combustor exit temperature (CET), trades between fan pressure ratio (FPR) and fan diameter, or bypass ratio (BPR), and exhaust type. Alternatives to this conventional gas turbine engine cycle, such as regenerative or compound cycles, were also addressed as possible gas generator configurations.

The first step in the cycle studies was to define the expected level of component and subsystem performance for the 2000 to 2010 time period. To do this, trends in engine overall efficiency, overall pressure ratio and combustor exit temperature were reviewed. Figure 4.1-1 shows how overall efficiency has progressed with time. Historical trends of overall pressure ratio and combustor exit temperature are shown in Figure 4.1-2. Pressure ratio, which has increased linearly for the last 30 years, is expected to continue the same trend. Turbine temperature was seen to increase linearly from 1940 to about 1970. But since about 1970, the rate has reduced and tended to flatten out. For commercial engines, there is no incentive to greatly increase the turbine temperature to improve thermal efficiency through the forecast period.

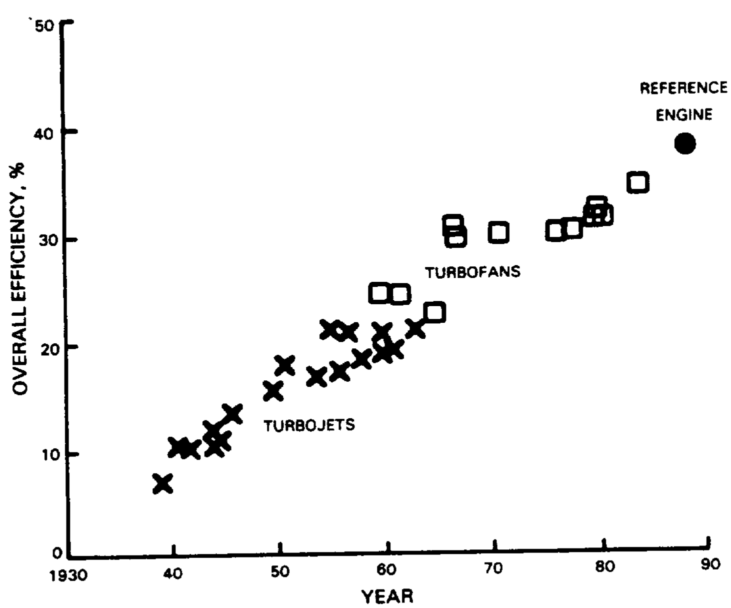

Figure 4.1-1 Benefit/Cost Study Reference Engine in Relationship to Historical Trends in Gas Turbine Engine Overall Efficiency

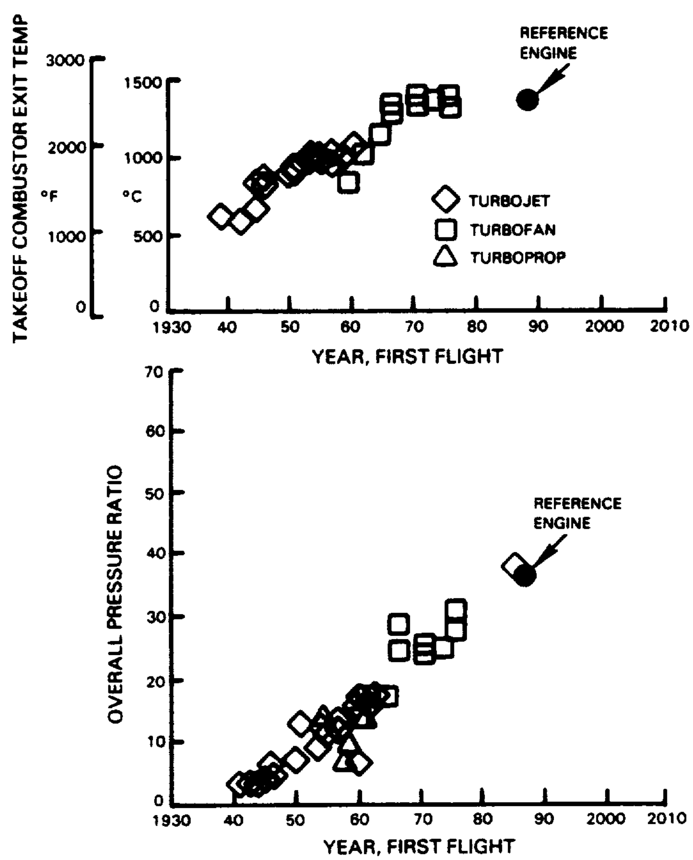

Figure 4.1-2 Benefit/Cost Study Reference Engine in Relationship to Historical Trends in Gas Turbine Engine Combustor Exit Temperature and Overall Pressure Ratio

There is expected to be a need for both small and large engines in the 2000 to 2010 time period. Obviously, the smaller engine, possibly a propfan, would be used to meet short haul airplane requirements while the large engine will remain the powerplant for long range, large airplanes. Therefore, projections of component efficiencies were made for both a large (266,892 N (60,000 lb) thrust) and a small (111,205 N (25,000 lb) thrust) engine. These are compared to the reference engine values in Table 4.1-I which shows that component efficiencies are expected to increase 1 to 2 percent in the forecast period. The difference between the large and small engines, about 1 percent, is due to scaling effects of Reynolds number and manufacturing limits. These values were used in the detailed cycle studies, and the benefit assessment discussed in Section 5.3.

The cycle studies, which examined variations in engine parameters and the effects on fuel consumption, were divided into two parts: parameters relating to thermal efficiency and those affecting propulsive efficiency; the product of the two is overall efficiency. Thermal efficiency is the effectiveness with which the gas generator converts the energy in the fuel into useful energy available for propulsion. Propulsive efficiency is the effectiveness of the conversion of this useful energy into actual propulsive power.

TABLE 4.1-I
COMPONENT EFFICIENCY PROJECTIONS

Component	Reference Engine Efficiency (%)	2010 Potential Efficiency (%) 111,205 N (25,000 lb) Thrust Engine	2010 Potential Efficiency (%) 266,892 N (60,000 lb) Thrust Engine
Fan and Low Pressure Compressor - polytropic	90	91.5	93
High Pressure Compressor - polytropic	91.6	92	93.1
High Pressure Turbine*	91.4	91.2 - 92.3	92.7 - 94.6
Low Pressure Turbine	92.5	93	94.4
Other			
Cooling and Leakage Air, (%)	18	8-12	6-10
Combustor Pressure Loss, (%)	4.4	3.0	3.0

* Variable with Turbine Cooling Air

4.1.1 Thermal Efficiency Evaluations

Thermal efficiency is determined by the compression ratio of the gas generator, combustor exit temperature, and the operating characteristics of the compressors, combustor and turbines involved in the conversion of heat energy to available propulsive energy. Technology advancements in component efficiencies, hot section materials and cooling capabilities have allowed the steady increases in overall pressure ratio and combustor exit temperature that were shown in Figure 4.1-2. Theoretical thermodynamics indicate that a continuation of the overall pressure ratio trend is desirable for better thermal efficiency, but that as component efficiencies are improved, the incentive for higher turbine temperatures is reduced. In the extreme case of components that are 100 percent efficient, i. e., no cooling or leakage flows or pressure losses, thermal efficiency of the ideal conventional cycle is penalized for higher combustor exit temperatures, as shown in Figure 4.1-3. This is caused by the adverse effects of real gas properties from the heat addition and turbine expansion processes.

Figure 4.1-4 presents trends of thermal efficiency with variations in overall pressure ratio and combustor exit temperature over a range of component efficiency levels. Turbine efficiencies are adiabatic and compressor efficiencies are polytropic. The figure shows that for any given level of overall pressure ratio, there occurs a corresponding combustor exit temperature for optimization of thermal efficiency. Increasing overall pressure ratio requires increased combustor exit temperature to maintain this optimization. Figure 4.1-4 further reveals that:

- o optimum turbine temperature decreases at constant overall pressure ratio as component efficiencies are improved;
- o with higher component efficiencies, there is more thrust specific fuel consumption incentive to increase overall pressure ratio;

- o there is an optimum overall pressure ratio beyond which thermal efficiency does not improve. However, this overall pressure ratio is well beyond any level of current interest in conventional gas turbine engine cycles.

Table 4.1-II presents projected turbine cooling air requirements for the 2000 to 2010 time period compared to the reference engine. These projections for various overall pressure ratios and combustor exit temperatures were made using the component efficiencies shown in Table 4.1-I. Advancements in materials and cooling technologies were assumed.

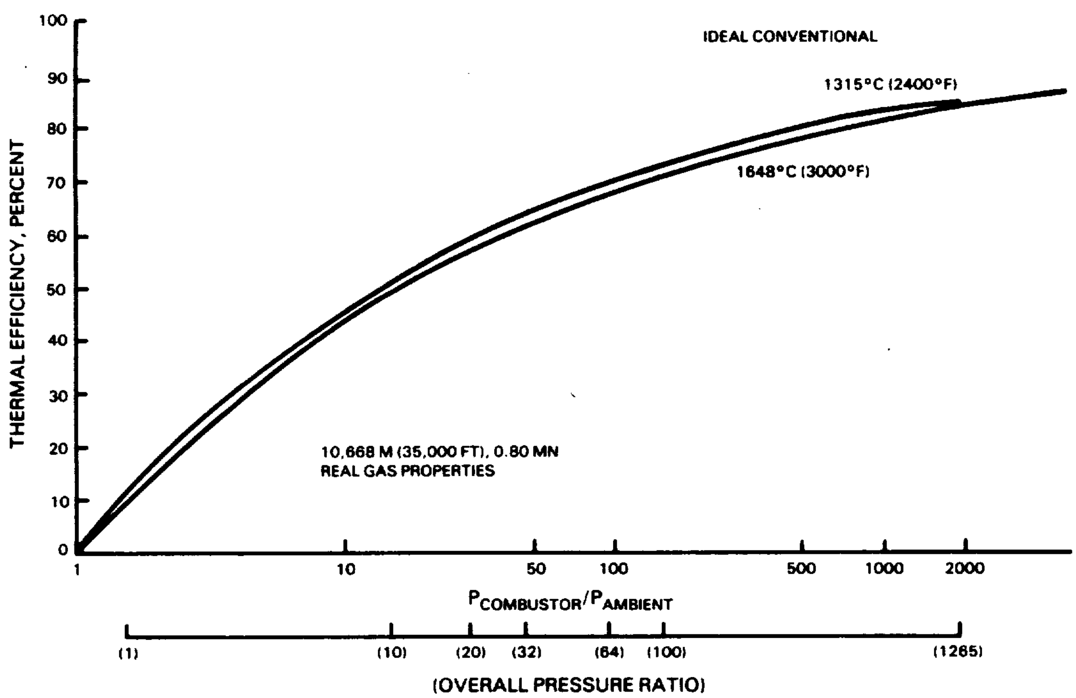

Figure 4.1-3 Thermal Efficiency of the Ideal Conventional Cycle Is Penalized for Higher Combustor Exit Temperatures

TABLE 4.1-II
PROJECTED TURBINE COOLING AIR REQUIREMENTS

High Pressure Turbine Location	Reference Engine 1435°C (2615°F) Max 38.6 OPR (%)	Year 2000 to 2010 1426°C (2600°F Max) 36 OPR (%)	Year 2000 to 2010 1648°C (3000°F) Max 72 OPR (%)
First Vane	6.4	2.9	5.7
First Vane Platform	1.0	0.3	0.7
First Blade	2.75	1.05	3.05
Second Vane	1.30	0.5	0.9
Second Blade	0.35	0.35	1.1
Secondary and Leakage Flow	6.3	3.0	3.0
Total	18.1	8.1	14.4

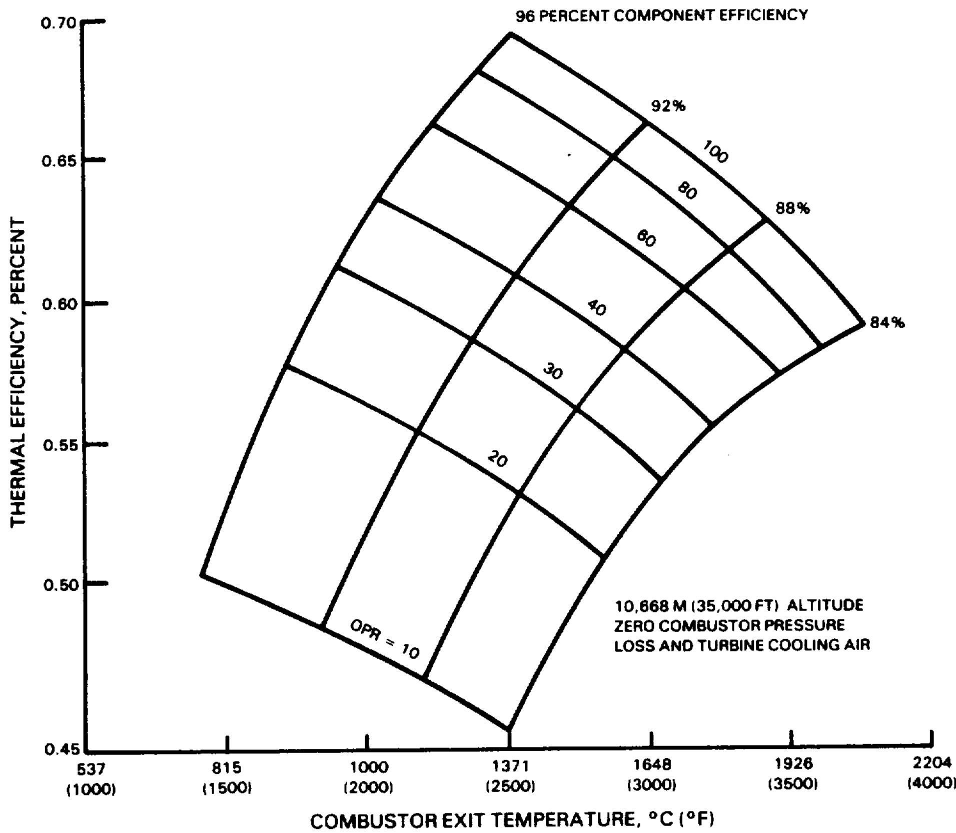

Figure 4.1-4 Turbine Temperatures for Maximum Thermal Efficiency at Varying Component Efficiency

As shown in Figure 4.1-5, with advancements in technology, less cooling air is required as overall pressure ratio increases. Incorporating the estimates from Figure 4.1-5 and Table 4.1-I into a combustor exit temperature/overall pressure ratio matrix results in the thrust specific fuel consumption pattern shown in Figure 4.1-6. Using this information, two high pressure compressor corrected exit flow sizes were evaluated, 1.24 and 2.4 kg/sec (2.75 and 5.5 lb/sec). The smaller size has reduced component efficiency levels due to minimum clearance effects on the smaller airfoils. The figure indicates a flattening of thrust specific fuel consumption improvement in the 60 to 70 overall pressure ratio range. Negligible incentive for higher combustor exit temperature is also shown and is most evident with the most efficient components (2.4 kg/sec (5.5 lb/sec) flow size). Table 4.1-III compares parameters of the conventional gas turbine engine operating cycle of the reference engine to those likely in the 2000 to 2010 time period.

Two alternate approaches to increase thermal efficiency outside of the conventional cycle were explored. Figure 4.1-7 illustrates, on an ideal basis, that regenerative cycles offer improvement at low overall pressure ratios, and compound turbo-diesel cycles offer ways of achieving very high cycle pressure ratios. Both options were evaluated assuming aggressive efficiency levels of the individual components.

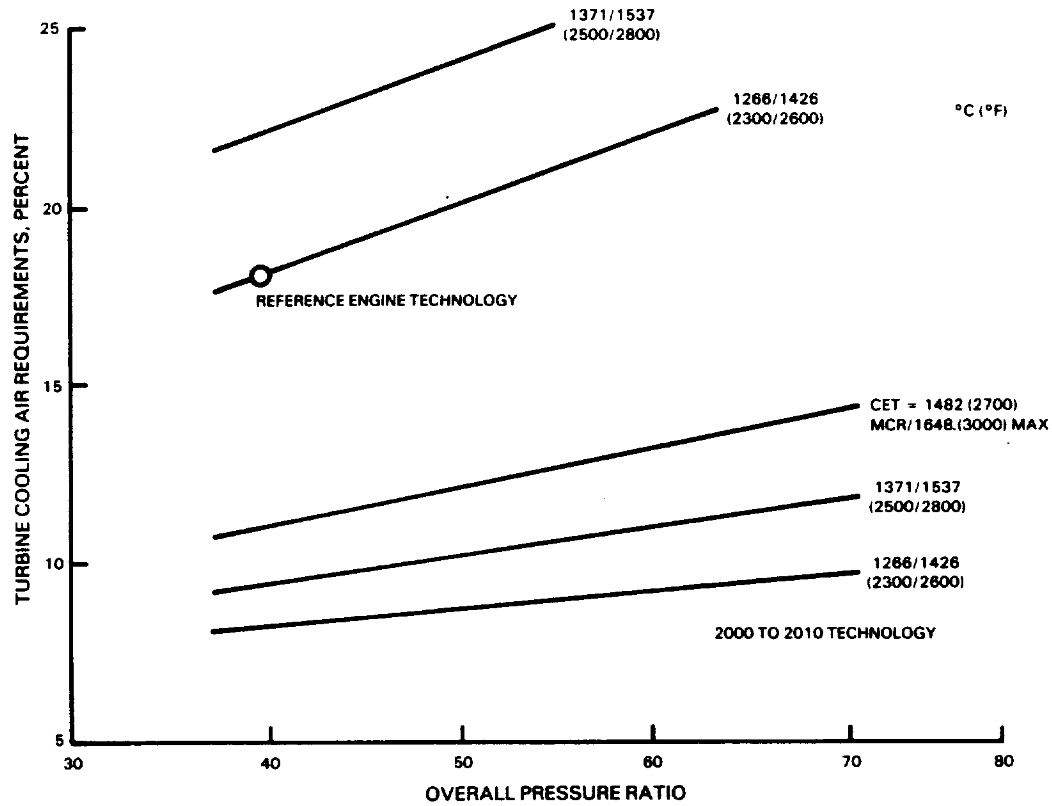

Figure 4.1-5 Turbine Cooling Air Trends

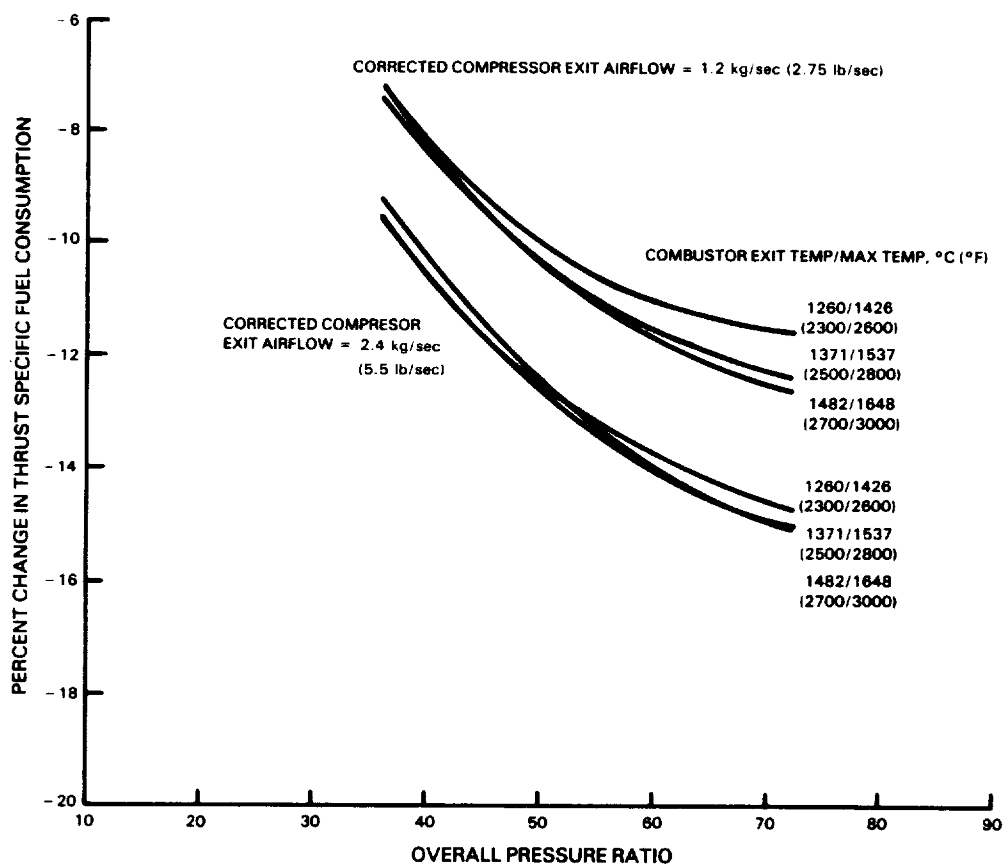

Figure 4.1-6 Reduction in Thrust Specific Fuel Consumption Compared to Reference Engine for Various Combustor Exit Temperatures and Overall Pressure Ratios

TABLE 4.1-III
COMPARISON OF CYCLE PARAMETERS

Parameter	Reference Engine	Year 2000 to 2010: 25,000 lb Thrust Engine	Year 2000 to 2010: 60,000 lb Thrust Engine
Overall Pressure Ratio	38.6	55	64
Combustor Exit Temperature °C (°F)	1268 (2315)	1329 (2425)	1329 (2425)
Thermal Efficiency (%)	0.581	0.619	0.638

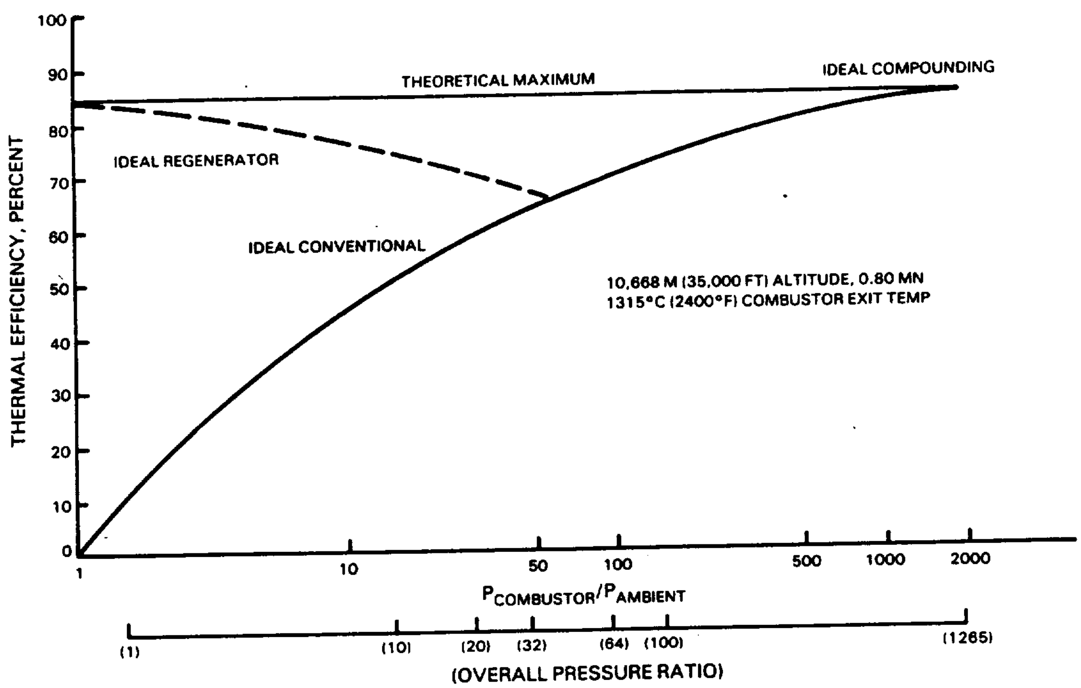

Figure 4.1-7 Effects of Regeneration and Cycle Compounding on Thermal Efficiency

Figure 4.1-8 shows the regenerative cycle results. As shown in the inset, in the regenerative cycle, heat is transferred via a heat exchanger from the rear of the engine to the flow exiting the high pressure compressor. This transferred heat could be extracted from between the high and low pressure turbines, or from behind the low pressure turbine. In both cases, aggressive levels of effectiveness, 90 percent, and pressure drop, 5 percent, on each side of the regenerator were assumed to give results which could be considered optimistic. By raising the combustor inlet temperature, less fuel addition is required for a given turbine temperature, and a thermal efficiency improvement is achieved similar to that with an overall pressure ratio increase. Thermal efficiencies peak at lower overall pressure ratios, showing some improvement over the conventional cycle. The disadvantage of the regenerative cycle is reduced thrust for a given gas turbine engine core size due to the energy extraction. To regain the decrease in thrust, a larger core size is required.

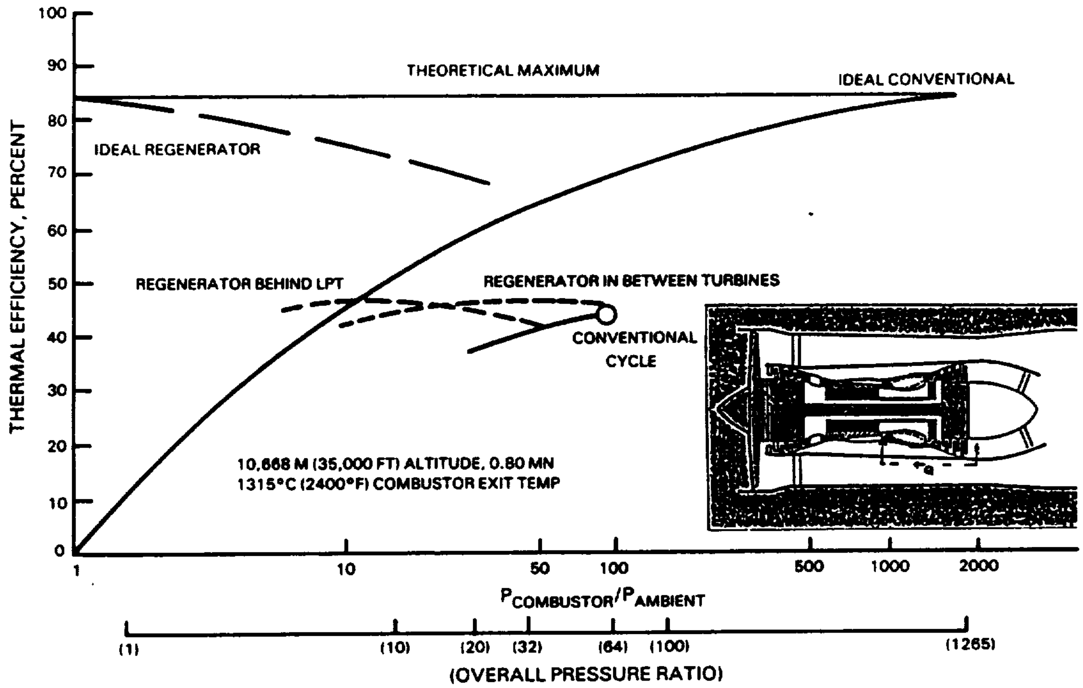

Figure 4.1-8 Results of Regenerator Cycle Studies Compared to Conventional Cycle Results

The weight penalty associated with larger gas generators coupled with the weight and complexity of efficient heat exchangers make the thermal efficiency gains from the regenerative cycle of minimal interest as an alternative for large commercial aircraft engines in the 2000 to 2010 time period.

The compound turbo-diesel cycle assumed a rotary diesel replacing the conventional combustor and feeding hot gas plus shaft power to the high pressure spool of an otherwise conventional two spool turbofan. High effective overall pressure ratios can be realized without the design concerns of very high pressures and temperatures of a typical combustor configuration. An evaluation was performed with the very aggressive assumptions for the diesel operating characteristics presented in Table 4.1-IV. The results of the evaluation, presented in Figure 4.1-9, did not show any thermal efficiency incentive over the conventional cycle.

TABLE 4.1-IV
COMPOUND TURBO-DIESEL ASSUMPTIONS

Operating Characteristic	Level
Diesel Compression Ratio	8
Pressure Loss (%)	5
Fuel Heating Value (Btu/lb)	19,260
Diesel Mechanical Efficiency (%)	95
Coolant Heat Loss (%)	15

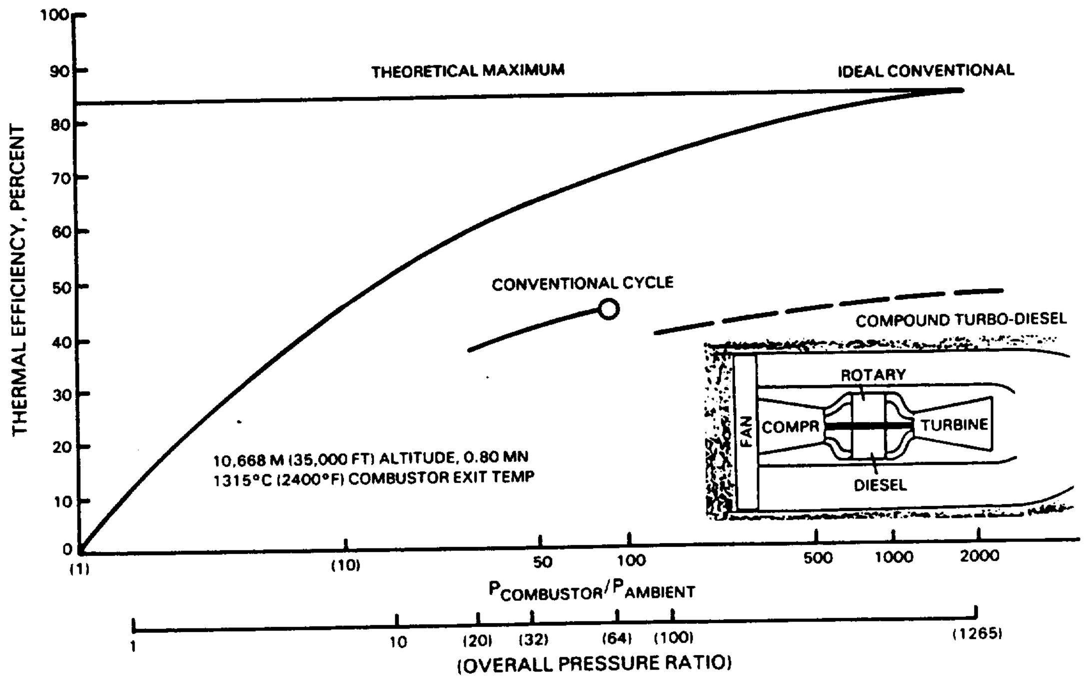

Figure 4.1-9 Results of Compound Turbo-Diesel Cycle Studies Compared to Conventional Cycle Results

4.1.2 Propulsive Efficiency Evaluations

The propulsive efficiency component of overall efficiency deals with the mechanisms of converting available energy to propulsive power. This not only includes the fan and the low pressure turbine, but any ducting and nozzle losses, and friction and pressure drags of the primary and fan stream cowling. The propulsive efficiency study concentrated primarily on trades of fan pressure ratio and bypass ratio, along with the exhaust system configuration (mixed or separate flow) and fan drive system (direct or geared).

Propulsive theory states that the higher the exhaust velocity relative to flight velocity, the greater the propulsive efficiency loss in an ideal component situation. This suggests that lower fan pressure ratio is desirable. However, there is a unique fan pressure ratio/bypass ratio combination which optimizes propulsive efficiency for any given amount of available energy, resulting in higher bypass ratios (increasing fan diameter) as fan pressure ratio is reduced. Figure 4.1-10 compares the loss mechanisms of the reference engine to those for two advanced engine configurations with different fan pressure ratios and incorporating projected advancements.

Presently, the compromise between fan and the low pressure turbine efficiencies operating at the same rotor speed tends to keep bypass ratio below 7 to maintain an acceptable fan diameter. Development of high efficiency reduction gear systems would allow larger diameter fans to run at optimum tip speeds for efficiency, which is lower than the speeds desired by the low pressure turbine for its peak efficiency.

Analysis of the separate exhaust versus mixed exhaust configuration is dependent upon the bypass ratio selection. In the intermediate bypass ratio range (5 to 10), an advantage for mixing the exhaust streams to reduce the velocity profile was evident. However, as shown in Figure 4.1-11, it becomes more difficult to efficiently perform the mixing process as the bypass ratio increases. Increasing pressure losses and decreased mixer efficiency change the emphasis from mixed flow to a separate exhaust nozzle system due to the the high bypass ratio cycle selection.

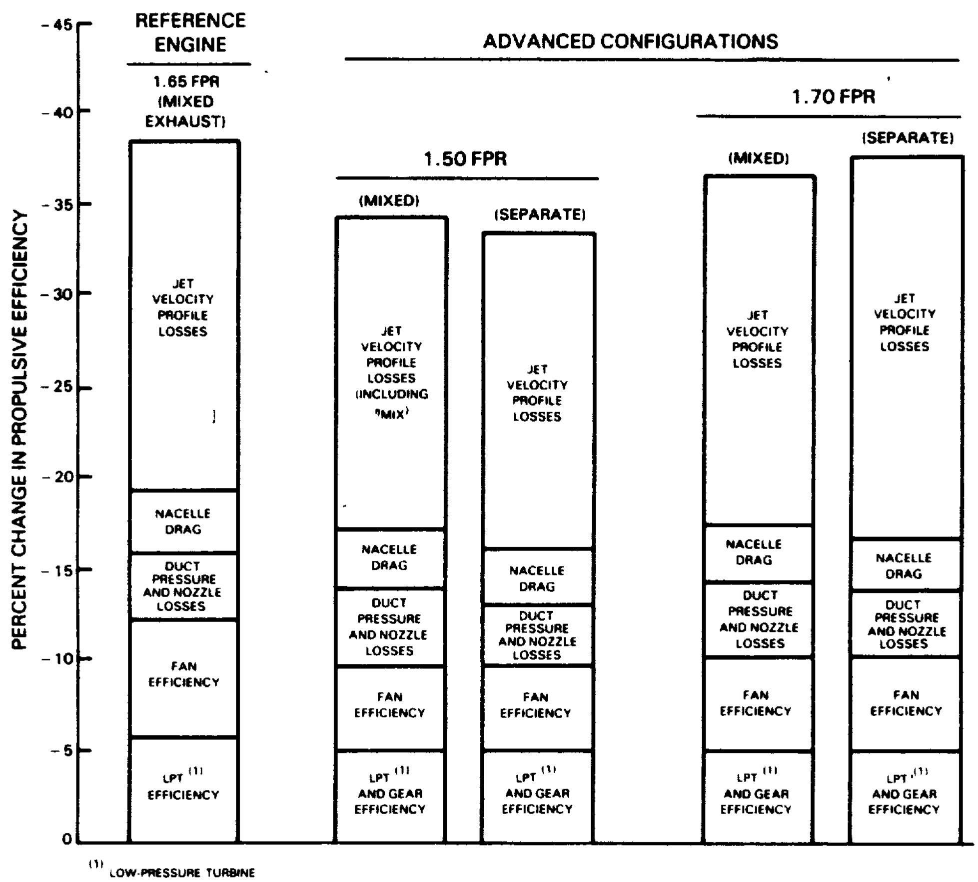

Figure 4.1-10 Comparison of Loss Mechanisms in Reference Engine to Those in Two Advanced Geared Engine Configurations

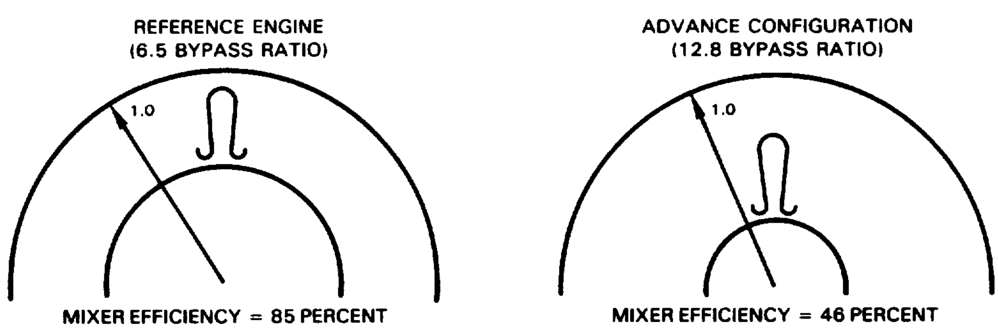

Figure 4.1-11 Comparison of Reference Engine Optimum Mixer Efficiency with High Bypass Ratio Optimum Mixer Efficiency

4.1.3 Noise Predictions

Evaluation of the acoustic impact of advanced technologies requires extension of current noise prediction methodology. The current turbomachinery noise prediction system is based on measured data from turbomachinery of known technologies. The current system combines fan/low pressure compressor and low pressure turbine noise and is scaled on bypass ratio.

The distribution of fan, compressor and turbine tone frequencies of engines using preliminary advanced technology candidates would be significantly different than those of current engines because of the different fan designs and operating speeds of the geared low pressure compressor and low pressure turbine. Turbomachinery noise predictions for an advanced cycle engine would require separating these tones from our data bases, correcting them in frequency and amplitude for the different component designs and operating conditions, and recombining them into an advanced technology prediction spectrum. Direct application of existing data bases without these changes would give erroneous predicted noise levels. Due to these problems, no attempts were made at noise prediction for the advanced cycles.

4.1.4 Selected Cycle Configurations

The cycle studies examined variations in engine parameters and the result on fuel consumption to determine overall efficiency. The first part, thermal efficiency, addressed selection of overall pressure ratio and combustor exit temperature. The second part, propulsive efficiency, addressed selection of fan pressure ratio, bypass ratio and exhaust type.

The thermal efficiency studies indicated that:

- there is a significant potential thrust specific fuel consumption reduction for increased overall pressure ratio,
- only moderate combustor exit temperature increases are required,
- there are no apparent advantages of alternative cycles to the conventional cycle.

The propulsive efficiency studies indicated that:

- reducing fan pressure ratio and increasing bypass ratio results in significant fuel efficiency gains;
- with high bypass ratio engines, there is no thrust specific fuel consumption benefit for mixed exhaust configurations over separate flow exhausts;
- advanced nacelle designs will reduce the thrust specific fuel consumption installation penalty for high bypass ratio engines.

4.1.5 Refinement of Technology Concepts

To achieve the projected efficiency requirements of the 2000 to 2010 time period with the selected cycle candidates, the preliminary technology concepts from the initial screening phase were reviewed and additional technology concepts were solicited. This effort resulted in the selection of nine primary advanced technology concepts that would be used in the flowpath studies mechanical feasibility studies. These concepts include:

for thermal efficiency advancement;
- o an advanced channel diffuser and combustor,
- o advanced diffuser/combustor materials,
- o a high efficiency high pressure turbine,
- o high efficiency compressors,
- o an advanced active clearance control,

for propulsive efficiency advancement;
- o a high efficiency low pressure turbine,
- o swept fan blades,
- o a geared low pressure spool,
- o an advanced nacelle.

A brief description of the specific advancements required from each of those technologies is presented below. Component improvements shown are due solely to the technology features noted. Effects due to variations in engine size or cycle are not included in this section but were accounted for in the benefits assessment discussed in Section 5.3. In that assessment, in order to account for component size differences between the reference engine and the advanced technology engines, the high-pressure turbine and high-pressure compressor efficiences shown in Tables 4.1-VI, 4.1-VII and 4.1-VIII were reduced by 0.6 percent and 0.3 percent, respectively.

4.1.5.1 Advanced Channel Diffuser and Combustor

The projected increase in overall engine pressure ratio results in an increase in the temperature of the air entering the diffuser/combustor and used for cooling turbine blades and vanes. Consequently, to meet the projected component efficiencies, an advanced diffuser/combustor will be needed to deliver higher quality cooling air with reduced pressure loss. Such a combustor would have to exhibit the characteristics identified in Table 4.1-V.

TABLE 4.1-V
DIFFUSER/COMBUSTOR TECHNOLOGY REQUIREMENTS

	Advanced Diffuser/Combustor
Pressure Loss, percent	
Diffuser	1.0
Liner	2.0
Pattern Factor	0.25
First Turbine Blade Temperature Profile, °C (°F) (Max to Average)	65 (150)
Turbine Cooling Air Temperature Reduction at Constant Overall Pressure Ratio, °C (°F)	
Combustor Inner Diameter Feed	-34 (-30)
Combustor Outer Diameter Feed	-23 (-10)

4.1.5.2 Advanced Diffuser/Combustor Materials

Advanced diffuser/combustor materials will be required to accommodate the higher temperature air entering the system. Liner segments will need a 148°C (300°F) increase in temperature capability to about 1204°C (2200°F). The diffuser case will need to be produced from castable/weldable high temperature alloys. The diffuser will probably have to incorporate the use of composites, possibly ceramic. Also, weight reductions will be achieved from shorter combustor lengths and use of lightweight sheet in liner segment support frames.

4.1.5.3 High Efficiency High Pressure Turbine

Advancements in high pressure turbine technology are required in both materials and aerodynamics. Advanced nickel materials will be needed to provide turbine disks 25 percent stronger than disks in the reference engine. Advanced single crystal superalloys for blades with prime reliable thermal barrier coatings will be needed to increase airfoil surface temperature capability 204 to 315°C (400 to 600°F). Likewise, a required 315°C (600°F) increase in vane temperature capability might be achieved with ceramics. In addition, a high pressure turbine case with a 204°C (400°F) higher temperature capability will be needed.

The aerodynamic advancements include the development of a three-dimensional design process which accounts for airfoil endwall losses. A 20 percent higher AN^2 (annulus area times rpm squared), 5 to 10 percent higher cooling effectiveness, and 0.063 cm (0.025 in) thinner airfoil trailing edges will also be required. In addition, improved clearance control will be needed to maintain tight running clearances. The increased high pressure turbine efficiency expected to result from these advancements is summarized in Table 4.1-VI.

TABLE 4.1-VI
ADVANCED HIGH PRESSURE TURBINE ADIABATIC EFFICIENCY IMPROVEMENTS

	Efficiency, percent
Reference Engine	91.4
Increased AN^2	+0.7
Reduced Trailing Edge Thickness	+0.2
Reduced Cooling Air (with ceramic vane)	+1.1 (+1.9)
Three-Dimensional Design Process	+1.0
Improved Clearance Control	+0.7
Total	95.1 (95.9)

4.1.5.4 High Efficiency Compressors

As in the high pressure turbine, compressor technology requires advancements in both materials and aerodynamics. Advanced aluminum blade alloys such as titanium-aluminide or forged aluminum blades bonded to a titanium compressor drum will be required. Advanced cast titanium cases will be needed to provide compressor cases 20 percent stronger than the case of the reference engine.

Required aerodynamic improvements include advanced controlled diffusion airfoils, endwall region improvements through the use of a three dimensional design process, tighter clearances from an improved clearance control system, and lower hub/tip radii. The increased axial compressor efficiency expected to result from these advancements is summarized in Table 4.1-VII. The results of similar advancements for axial-centrifugal compressors is presented in Table 4.1-VIII.

TABLE 4.1-VII
ADVANCED AXIAL COMPRESSOR POLYTROPIC EFFICIENCY IMPROVEMENTS

	Efficiency, percent
Reference Engine	91.6
Advanced Controlled Diffusion Airfoils	+0.5
Three-Dimensional Design Process	+0.5
Reduced Hub/Tip Radii	+0.2
Improved Clearance Control	+0.6
Total	93.4

TABLE 4.1-VIII
ADVANCED AXIAL-CENTRIFUGAL COMPRESSOR POLYTROPIC EFFICIENCY IMPROVEMENTS

	Efficiency, percent
Reference Engine	91.6
Axial Stage Advancements	+1.6 to +1.8
Centrifugal Stage Advancements	+2.0
Average Advancements	+1.9
Axial to Axial-Centrifugal	-1.2
Total	92.3

4.1.5.5 Advanced Active Clearance Control

The final thermal efficiency advancement expected by the 2000 to 2010 time period is active clearance control technology. To achieve the necessary component efficiency improvement, tighter running clearances (10 mils) will be required in the compressor and turbine areas. Closed loop active clearance control will be required to achieve those clearances. In such a closed loop system, clearances will be continuously measured and adjusted for optimum performance during steady state operation. For transient operation in the closed loop system, a lightweight mechanical actuation device may be required to quickly open clearances and avoided pinching.

4.1.5.6 High Efficiency Low Pressure Turbine

Propulsive efficiency improvement will be obtained from low pressure turbine advancements. Required materials advancements for this component include a low expansion alloy case and titanium-aluminide blading in the rear two stages. Aerodynamic advancements include the three-dimensional airfoil design process and an AN^2 approximately 300 percent higher than in the reference engine. The increased low pressure turbine efficiency expected to result from these advancements is summarized in Table 4.1-IX.

TABLE 4.1-IX
ADVANCED LOW PRESSURE TURBINE ADIABATIC EFFICIENCY IMPROVEMENTS

	Efficiency, percent
Reference Engine	92.5
Increased AN^2	+1.1
Three-Dimensional Design Process	+0.5
Improved Clearance Control	+0.3
Total	94.4

4.1.5.7 Swept Fan Blades

Additional propulsive efficiency improvements will be obtained from swept fan blade technology. Axially slanting the leading edges of fan blades reduces the relative velocity of the airstream on the leading edge of the blade. In addition to sweeping the blade, aerodynamic benefits are expected to accrue from use of an advanced three-dimensional design process. Elimination of the fan blade shroud and lighter hollow, titanium alloy blades will also improve efficiency. The fan disk will also be made of lighter alloys with composite reinforcement.

The efficiency of the reference engine fan with the conventional, shrouded blade is 87.2 percent operating at a fan pressure ratio of 1.65. It is expected that fan efficiency with the advanced swept fan blade design at a 1.5 fan pressure ratio will be 91.5 percent and decrease to 90.2 percent at a fan pressure ratio of 1.7.

4.1.5.8 Geared Low Pressure Spool

The efficiency improvements expected in the high speed, low pressure turbine and in the fan are dependent on development of a highly efficient, geared, low pressure spool. This geared spool would enable the low pressure turbine to achieve the required high levels of AN^2 and, at the same time, allow the fan to operate at lower fan pressure ratio and higher bypass ratio.

To achieve high efficiency, the gears will need to exhibit fatigue strength 40 percent greater than today's gears and will probably be made of advanced rapid solidification rate powders. Bearings will also incorporate advanced metallurgical concepts. Lubricants will have up to 3413.8 kg/cm (3000 lb/in) greater load carrying capability and a flash point of about 204°C (400°F), 51°C (125°F) higher than lubricants in the reference engine. The gear housing will have to be made of cast aluminum or composites for strength and light weight. These advances should provide a gear for the low pressure spool that operates at about 99.3 percent efficiency, thereby maintaining the advantages of both the fan and the high speed, low pressure turbine.

4.1.5.9 Advanced Nacelle

Propulsive efficiency designed into an engine can only be realized if its installation does not penalize performance with drag or weight. The nacelle of the 2000 to 2010 time period will be wrapped around a higher bypass ratio engine than the reference engine and, therefore, must include many advancements to minimize diameter and associated drag penalties. Advanced stiffening techniques will have to be included for a nacelle on a high speed, more flexible core. The nacelle inlet, fan cowl, fan nozzle, fan discharge and fan reverser will probably all incorporate composites for weight reduction.

4.2 FLOWPATH DEFINITION

Based on the required technologies identified in the cycle studies, flowpath studies were conducted to produce gaspath geometry for the mechanical feasibility studies. In this flowpath definition phase, several potential engine flowpath candidates were evaluated before selection of three final flowpaths.

The flowpath definition phase is described in three subsections. Subsection 4.2.1 presents an overview encompassing significant results from the cycle studies. A description of the evaluation of the eight candidate flowpaths is presented in subsection 4.2.2. The three final flowpaths for undergoing mechanical feasibility studies are identified in subsection 4.2.3.

4.2.1 Overview

The results of the cycle studies indicate trends toward improved fuel consumption through higher overall pressure ratio and higher bypass ratio/lower fan pressure ratio. The combination of high bypass ratio/low fan pressure ratio presents a low pressure spool design problem. Low fan pressure ratio implies low tip speed (for optimal fan performance) and high bypass ratio implies large diameter. The coalescence of low tip speed and large diameter results in slow low pressure rotor speed. This, in turn, compromises both low pressure compressor and low pressure turbine performance. In addressing this problem, both direct drive and geared arrangements were assessed to identify the potential advantages and disadvantages of each. Based on these considerations, the eight cycle candidates presented in Table 4.2-I were selected for evaluation in the flowpath definition studies. To ensure that all potentially benefitting technologies were considered, both two spool and three spool configurations were selected.

TABLE 4.2-I
CYCLE CANDIDATES FOR FLOWPATH DEFINITION

Candidate	Bypass Ratio	Fan Pressure Ratio	Overall Pressure Ratio	Takeoff Thrust N (lb)	Spools/ Configuration
Reference Engine	7.20	1.65	38.6	173,479 (39,000)	2/Dir. Drv.
1	12.8	1.50	64.0	266,892 (60,000)	2/Geared
2	13.1	1.50	64.0	266,892 (60,000)	3/Geared (no low compressor)
3	13.1	1.50	64.0	266,892 (60,000)	3/Geared
4	21.0	1.30	64.0	266,892 (60,000)	2/Geared
5	12.3	1.50	46.0	111,205 (25,000)	2/Geared
6	11.7	1.50	55.0	111,205 (25,000)	2/Geared
7	11.9	1.50	64.0	266,892 (60,000)	2/Dir. Drv.
8	9.00	1.70	64.0	266,892 (60,000)	2/Dir. Drv.

The question of pressure split for the high overall pressure ratio (64) was also studied. In the case of the two spool configurations, several pressure splits were evaluated. Figure 4.2-1 shows how high pressure turbine AN^2 varies with the pressure ratio of the high pressure compressor for the inlet conditions given. As pressure ratio is transferred onto the high pressure rotor, the high pressure compressor inlet corrected flow increases, resulting in larger inlet annulus area (for constant inlet specific flow) and hence, larger diameter (for constant inlet hub/tip ratio). For a given high pressure compressor inlet corrected tip speed, this results in lower high pressure rotor speed. This, combined with the fact that the high pressure turbine work requirement increases as pressure ratio is transferred onto the high pressure spool, causes excessive gas turning if the turbine velocity ratio is held constant. This turning can be reduced in either of three ways: (1) increased velocity ratio, (2) decreased AN^2 or (3) lower high spool pressure ratio.

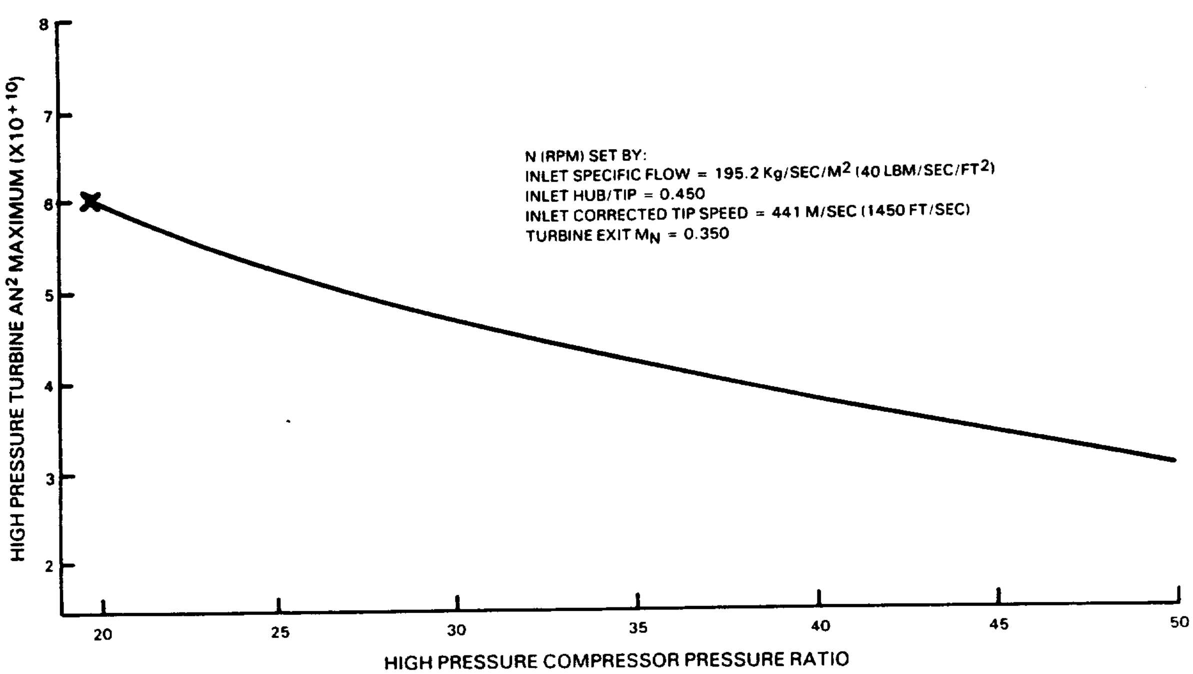

Figure 4.2-1 Effect of Increasing High Pressure Compressor Pressure Ratio on High Pressure Turbine AN^2

Increasing velocity ratio was deemed unacceptable since a 0.65 level was already being used. Decreasing AN^2 was viewed as unfavorable because of its associated loss in turbine performance. Consequently, it was decided to lower the pressure ratio of the high pressure spool to a level which satisfied both the high pressure compressor and high pressure turbine.

Initially, it was believed that the three spool arrangement would not require a low pressure compressor. However, the intermediate pressure compressor requires a level of pressure sufficient to allow both the compressor (specific flow, hub/tip ratio and tip speed) and turbine (velocity ratio and AN^2) design criteria to be met. As a result, low pressure compressor staging was required.

The pressure split between the high and intermediate spools in the three spool configuration also needed to be determined. A major consideration in that decision was that even though the flowpath was being configured as an all axial compression system, there was an alternate approach which would replace some of the axial stages on the high spool with a centrifugal stage. Mechanical tip speed, metal temperature and pressure ratio were items of concern in the design of the centrifugal stage. Consequently, the final split of 2.6 x 4.92 x 5.0 was selected, permitting an acceptable centrifugal compressor design.

4.2.2 Evaluation of Candidate Flowpaths

This section presents a component-by-component description of the eight cycle candidate flowpaths. For comparison purposes, the flowpath of the reference engine is presented in Figure 4.2-2 and the flowpaths of candidates 1 through 8 are presented in Figures 4.2-3 through 4.2-10, respectively.

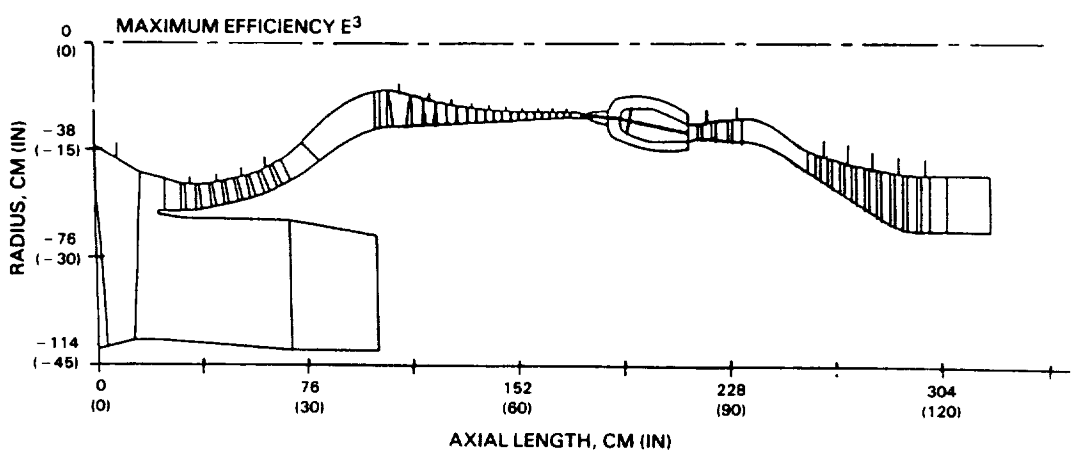

Figure 4.2-2 Flowpath of Reference Engine

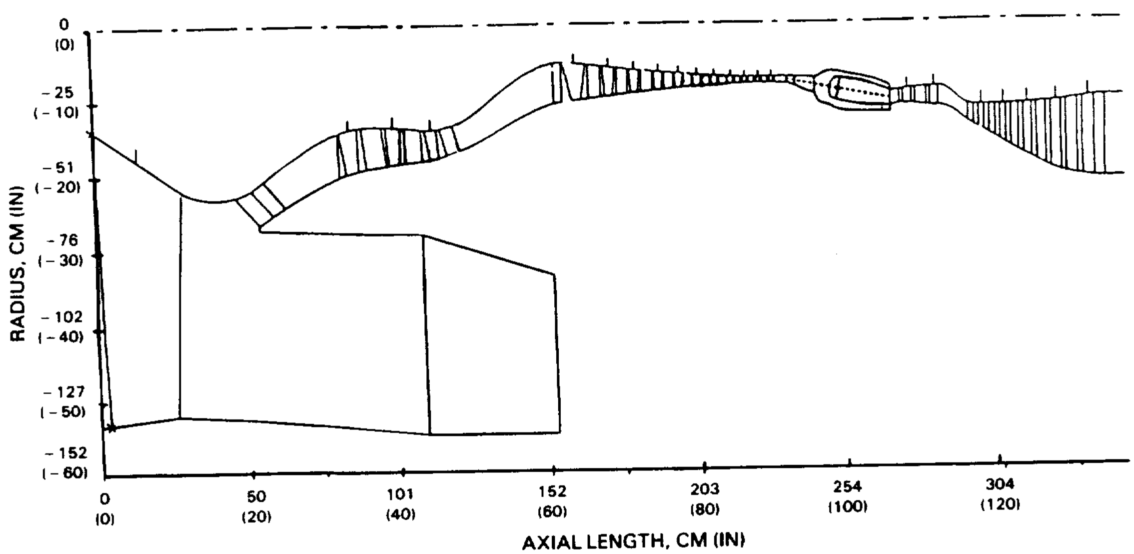

Figure 4.2-3 Flowpath of Advanced Technology Turbofan Candidate 1

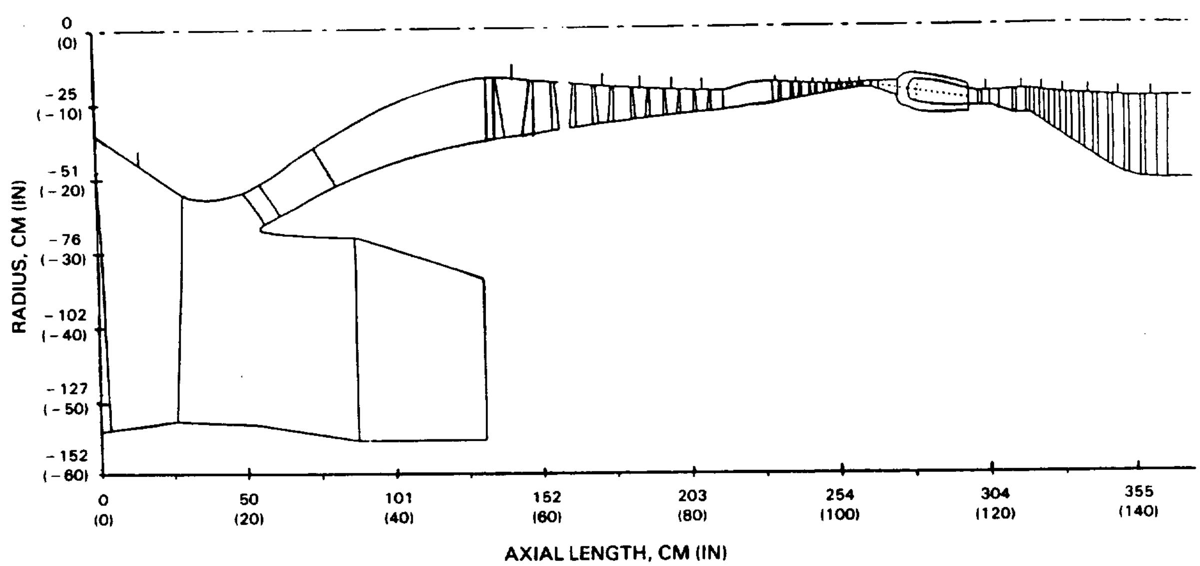

Figure 4.2-4 Flowpath of Advanced Technology Turbofan Candidate 2

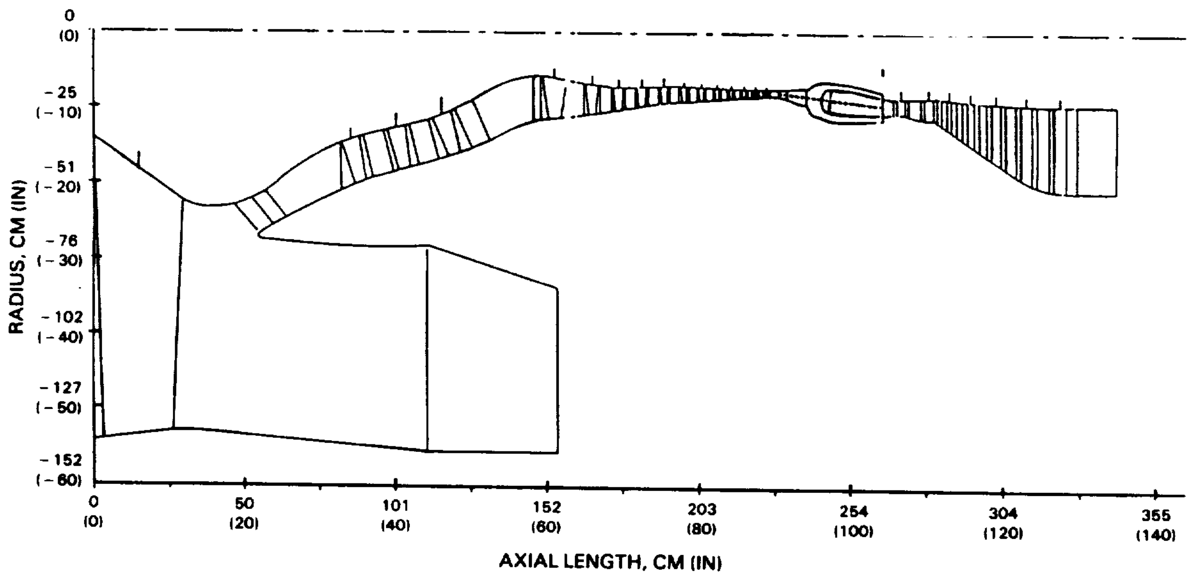

Figure 4.2-5 Flowpath of Advanced Technology Turbofan Candidate 3

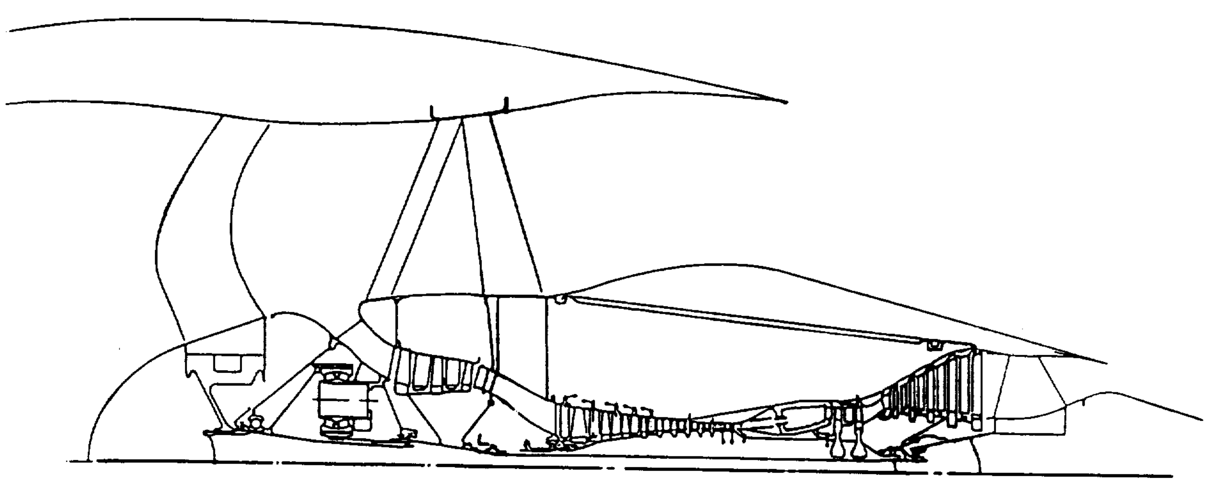

Figure 4.2-6 Flowpath of Advanced Technology Turbofan Candidate 4

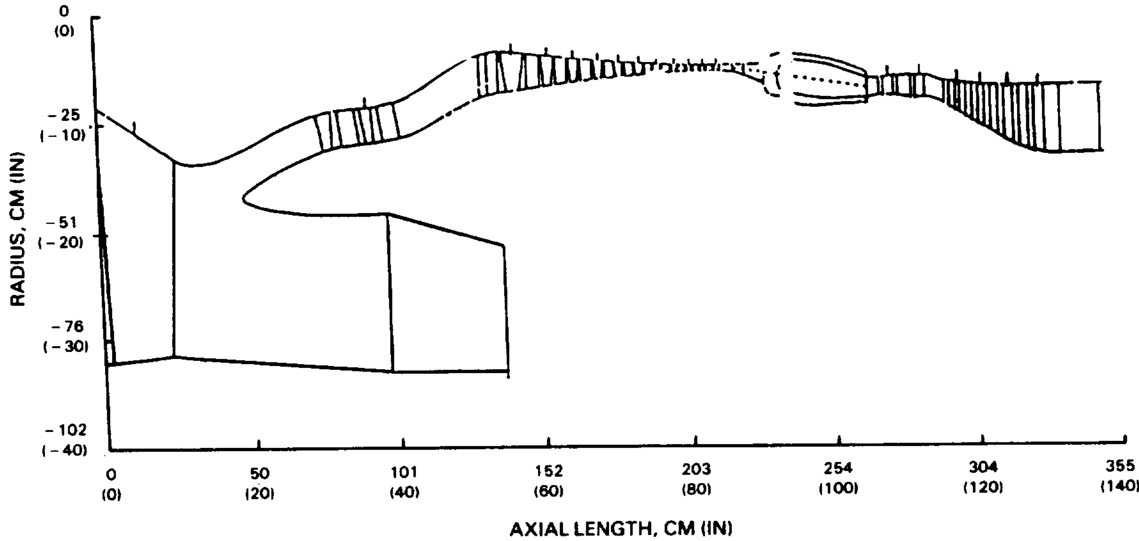

Figure 4.2-7 Flowpath of Advanced Technology Turbofan Candidate 5

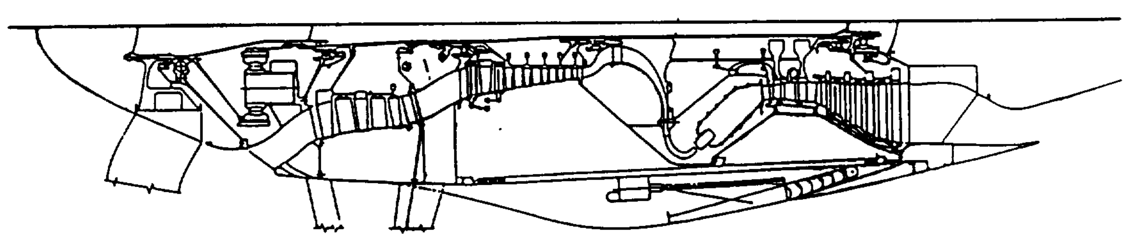

Figure 4.2-8 Flowpath of Advanced Technology Turbofan Candidate 6

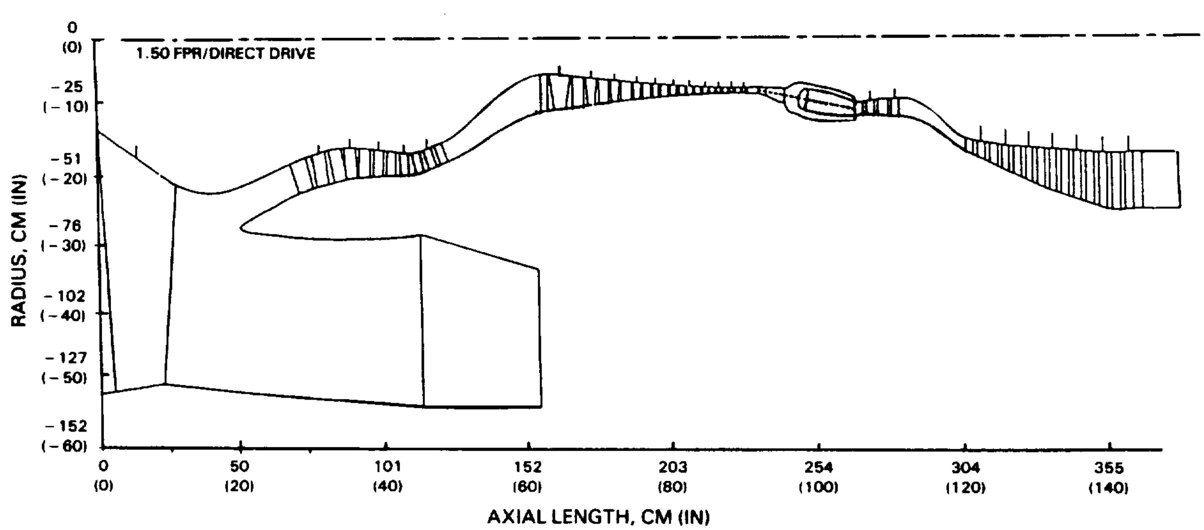

Figure 4.2-9 Flowpath of Advanced Technology Turbofan Candidate 7

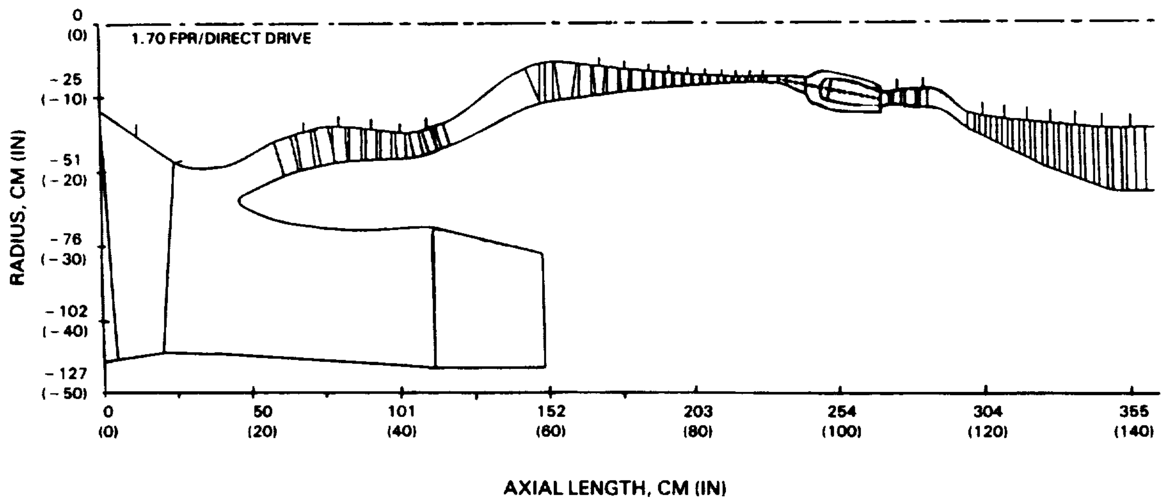

Figure 4.2-10 Flowpath of Advanced Technology Turbofan Candidate 8

4.2.2.1 Fan

The fan design of both the two and the three spool configurations was substantially different from that of the reference engine. All candidates incorporated low hub/tip ratio (0.260), low tip speed (with the exception of direct drive configurations) and high specific flow 219.6 kg/sec/m^2 (45.0 lbm/sec/ft^2) design criteria. Table 4.2-II compares the fan flowpath designs of the eight candidates to the reference engine.

TABLE 4.2-II
SUMMARY OF FLOWPATH CANDIDATE FAN DESIGNS

		Candidate							
		1	2	3	4	5	6	7	8
	Reference Engine	1.5 FPR 2 Spool	3 Spool (No LPC)	3 Spool (With LPC)	1.3 FPR 2 Spool	2 Spool 46 OPR	2 Spool Axi-Cent.	2 Spool Dir Drv	2 Spool Dir Drv
Corrected Flow	1498.0	2612.0	2664.0	2664.0	4167.0	940.8	1052.0	2437.0	1883.0
Outer Diameter Pressure Ratio	1.65	1.50	1.50	1.50	1.30	1.50	1.50	1.50	1.70
Inner Diameter Pressure Ratio	1.50	1.27	1.29	1.27	1.10	1.27	1.28	1.37	1.45
Bypass Ratio	7.2	12.8	13.1	13.1	21.0	12.3	11.7	11.9	8.96
Tip Diameter, cm (in)	215.9	271.2	274.0	274.0	341.8	162.8	172.2	262.1	230.4
	(85.0)	(106.8)	(107.9)	(107.9)	(134.6)	(64.1)	(67.8)	(103.2)	(90.71)
Configuration	Direct	Geared	Geared	Geared	Geared	Geared	Geared	Direct	Direct
Corrected Tip Speed, m/sec	441	356	356	356	274	356	356	441	441
(ft/sec)	(1450)	(1170)	(1170)	(1170)	(900)	(1170)	(1170)	(1450)	(1450)
Flow per Unit Area, lbm/sec/ft^2	43.0	45.0	45.0	45.0	45.0	45.0	45.0	45.0	45.0
(kg/sec/m^2)	(209.8)	(219.6)	(219.6)	(219.6)	(219.6)	(219.6)	(219.6)	(219.6)	(219.6)
Hub/Tip Ratio	0.340	0.260	0.260	0.260	0.260	0.260	0.260	0.260	0.260
Blade Aspect Ratio	4.00	3.00	3.00	3.00	3.00	3.00	3.00	3.00	3.00
Number of Blades	36	24	24	24	24	24	24	24	24

LPC = Low Pressure Compressor

The inner diameter design of the fan stator in the eight candidates was radically different from that of the reference engine. The lower radial elevation and axial placement further downstream relative to the fan rotor allows the majority of dirt to be centrifuged into the fan duct stream, thereby significantly reducing deterioration of the core.

Since the cycle study results showed incentive for a lower fan pressure ratio/higher bypass ratio, a 1.3 fan pressure ratio/21.0 bypass ratio cycle (candidate 4) was analyzed. The high bypass ratio resulted in a 342 cm (135 in) fan diameter which would cause severe installation penalties and erode all the fuel consumption benefits derived from the cycle. For this reason, no further analysis was conducted on candidate 4.

The fan tip speed for the two direct drive cycles was selected by trading fan and low pressure turbine performance as a function of rpm. Removal of the part span shroud and sweeping the blade significantly reduced the penalty associated with high tip speed fan designs. This, in conjunction with the fact that the low pressure turbine performance improves with increasing speed (higher velocity ratio), resulted in the determination that both the 1.50 and the 1.70 fan pressure ratio direct drive cycles would optimize at the same fan corrected tip speed of 441 m/sec (1450 ft/sec).

4.2.2.2 Low Pressure Compressor

Except for candidates 2 and 5, the pressure ratio requirements of the low pressure compressor were higher than those of the reference engine. Table 4.2-III compares the low pressure compressor flowpath designs of the eight candidates to the reference engine.

Initially, it was believed that the low pressure compressor would operate at the same rotor speed as the fan for the geared arrangements. However, this necessitated 5 and/or 6 stages (as is evidenced by the direct drive configurations) to obtain acceptable levels of performance and aerodynamic loadings. Since this was unacceptable, an alternate approach was considered whereby the low pressure compressor would run at the same speed as the low pressure turbine. This reduced the required number of stages substantially. The higher tip speeds of the candidates relative to the reference engine allowed higher pressure ratio per stage.

4.2.2.3 Intermediate Case

The length of the reference engine's intermediate case was set by an aerodynamic loading level commensurate with a late 1980s time frame. The later certification date for the flowpath candidates (2010) permits a higher loading parameter. Table 4.2-IV compares the intermediate cases of the eight candidates to the reference engine.

TABLE 4.2-III
SUMMARY OF FLOWPATH CANDIDATE LOW PRESSURE COMPRESSOR DESIGNS

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR	6 2 Spool Axi-Cent.	7 2 Spl DD 1.5 FPR	8 2 Spl DD 1.7 FPR
Pressure Ratio	1.84	2.52	N/A	2.05	2.909	1.795	2.15	2.33	2.21
Number of Stages	4	2		2	2	2	3	5	5
Configuration	Fan-Tied	LPT-Tied		------------------LPT-Tied ----------------				Fan-Tied	Fan-Tied
Average Aspect Ratio	2.30	1.90		1.90	1.90	1.90	1.91	2.60	2.60
Average Gap/Chord Ratio	0.930	1.00		1.00	0.890	0.938	0.992	0.997	1.10
Axial Velocity/Wheel Speed	0.860	0.524		0.522	0.426	0.453	0.497	0.535	0.535
Flow/Unit Area, $lbm/sec/ft^2$	36.0	35.5		35.5	35.5	35.5	35.5	35.5	35.5
($kg/sec/m^2$)	(175.7)	(173.2)		(173.2)	(173.2)	(173.2)	(173.2)	(173.2)	(173.2)
Exit Mach Number	0.430	0.460		0.420	0.485	0.350	0.385	0.535	0.535
Exit Swirl Angle	0°(Axial)	0°(Axial)		0°(Axial)	0°(Axial)	0°(Axial)	0°(Axial)	0°(Axial)	0°(Axial)
Average Diffusion Factor	0.370	0.464		0.447	0.469	0.460	0.445	0.455	0.454
Average Endwall Loading	0.290	0.347		0.367	0.396	0.390	0.371	0.276	0.264
Number of Airfoils	764	253		186	349	168	224	813	613

TABLE 4.2-IV
SUMMARY OF CANDIDATE FLOWPATH INTERMEDIATE CASE DESIGNS

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR	6 2 Spool Axi-Cent.	7 2 Spl DD 1.5 FPR	8 2 Spl DD 1.7 FPR
Length, cm (in)	39.62	37.5	54.6	23.62	39.14	25.19	18.49	40.1	38.3
	(15.60)	(14.8)	(21.5)	(9.30)	(15.41)	(9.92)	(7.28)	(15.8)	(15.1)
Strut Axial Chord, cm (in)	27.4	--	--	--	--	--	--	--	--
	(10.8)								
Loading Parameter $\Delta P_s/Q$	0.40	0.60	0.60	0.60	0.60	0.60	0.60	0.60	0.60
Inner Diameter Radius, cm (in)	24.89	21.00	20.04	8.20	26.67	11.63	7.62	25.65	21.10
	(9.80)	(8.27)	(7.89)	(3.23)	(10.50)	(4.58)	(3.00)	(10.1)	(8.31)

Length measured from low pressure compressor exit stator trailing edge to high pressure compressor first rotor leading edge.

4.2.2.4 Intermediate Pressure Compressor

The intermediate pressure compressors of the three spool configurations, candidates 2 and 3, incorporated high inlet specific flow, low hub/tip ratio, high inlet corrected tip speed. The arrangement which retained the low pressure compressor staging was a 5 stage, 4.92 pressure ratio intermediate pressure compressor. The three spool, which did not have any low pressure compressor staging, resulted in a 6 stage, 5.8 pressure ratio intermediate pressure compressor. The flowpaths of candidates 2 and 3 are illustrated in Figures 4.2-3 and 4.2-4, respectively. Table 4.2-V compares the intermediate pressure compressor flowpaths of the candidates to the reference engine.

4.2.2.5 High Pressure Compressor

Numerous pressure ratio combinations for the low pressure spool and high pressure spool were analyzed. The majority of this effort centered on candidate 1 and the results were applied to the other candidates.

TABLE 4.2-V
SUMMARY OF FLOWPATH CANDIDATE INTERMEDIATE PRESSURE COMPRESSOR DESIGNS

	Reference Engine	Candidate 1: 1.5 FPR 2 Spool	Candidate 2: 3 Spool (No LPC)	Candidate 3: 3 Spool (With LPC)	Candidate 4: 1.3 FPR 2 Spool	Candidate 5: 2 Spool 46 OPR
Pressure Ratio	N/A	N/A	5.8	4.92	N/A	N/A
Inlet Corrected Flow,			68.7	38.2		
kg/sec (lb/sec) $W\sqrt{\theta}/\delta$			(151.5)	(84.3)		
Corrected Tip Speed, m/sec			442	441		
(ft/sec)			(1452)	(1450)		
Number of Stages			6	5		
Inlet Hub/Tip Ratio			0.440	0.490		
Exit Hub/Tip Ratio			0.768	0.760		
Average Aspect Ratio			1.90	1.50		
Average Gap/Chord Ratio			0.975	1.10		
Flow Coefficient			0.651	0.614		
Flow/Unit Area, $lbm/sec/ft^2$			40.0	40.0		
($kg/sec/m^2$)			(195.2)	(195.2)		
Exit Mach Number			0.435	0.405		
Reaction Average			0.660	0.700		
Average Diffusion Factor			0.446	0.468		
Average Airfoil Row Loading			0.357	0.347		

Initially, a preliminary pressure ratio split of 2.56 x 25.0 was assessed. The rotor speed determined by the high pressure turbine AN^2 level of 6.0×10^{10} was 18,000 rpm. This speed yielded a compressor inlet tip speed of 493 m/sec (1620 ft/sec) assuming the 0.45 inlet hub/tip ratio and the 195.2 $kg/sec/m^2$ (40.0 $lbm/sec/ft^2$) inlet specific flow. This speed was judged to be mechanically unfeasible.

An obvious solution would be to slow down the high pressure rotor which would result in a high pressure turbine with:

1. a reduced velocity ratio (reduced efficiency); or

2. a constant velocity ratio and increased elevation, both of which would increase weight and decrease performance.

Since neither of these alternatives was attractive, a study was undertaken to determine the most favorable pressure split.

For a 111,205 N (25,000 lb) thrust size configuration, an all axial and an axial-centrifugal high pressure compressor were evaluated. Work was initiated on a 46 overall pressure ratio cycle (candidate 5) which was estimated based on some early cycle work. A 55 overall pressure ratio cycle (candidate 6) was ultimately selected based on more detailed analysis. The majority of the analysis centered around this cycle.

Interest in an axial-centrifugal high pressure compressor developed because of the small high pressure compressor exit corrected flow size 13.4 kg/sec/m^2 (2.75 lb/sec/ft^2) and, hence, small blade and vane spans. Also, a centrifugal compressor offered a shorter, stiffer high pressure rotor which is structurally beneficial in high speed designs. The selected configuration, shown in Figure 4.2-11, was a 6 stage axial (6:1 pressure ratio), 1 stage centrifugal (3.34:1 pressure ratio) high pressure compressor. Table 4.2-VI summarizes the characteristics of the axial-centrifugal high pressure compressor.

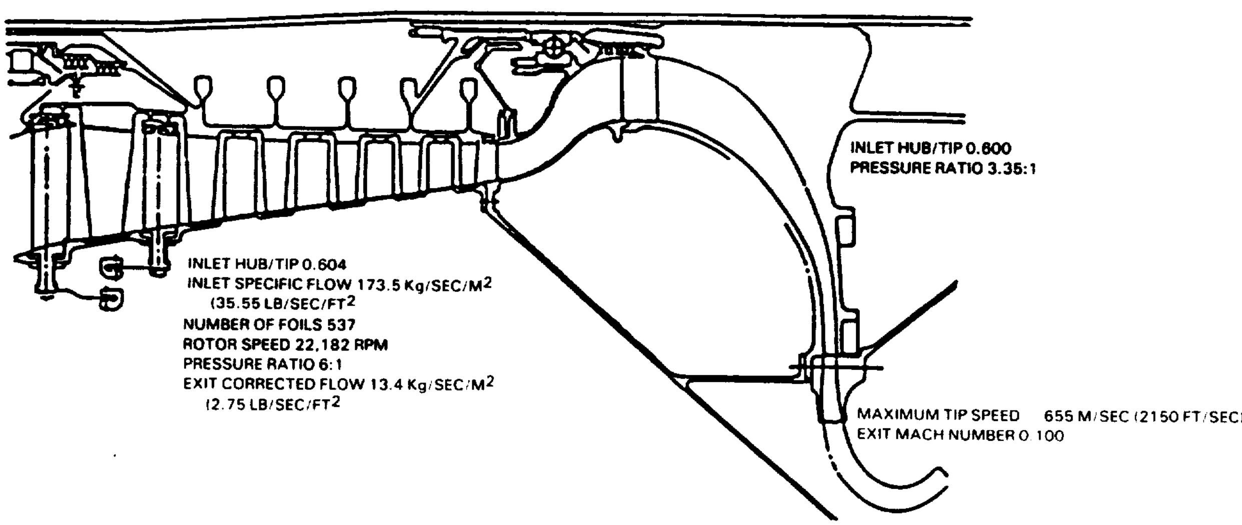

Figure 4.2-11 Axial-Centrifugal High Pressure Compressor of Candidate 6

TABLE 4.2-VI
AXIAL-CENTRIFUGAL HIGH PRESSURE COMPRESSOR SUMMARY

Inlet Temperature, °K (°R)	593 (1069)
Pressure Ratio	3.34
Inlet Weight Flow, kg/sec/m^2 (lb/sec/ft^2)	37.5 (7.70)
Specific Speed	72.5
Exit Corrected Flow, kg/sec/m^2 (lb/sec/ft^2)	13.42 (2.75)
Mechanical Tip Speed, m/sec (ft/sec)	621 (2040)
Tip Diameter, cm (in)	56.13 (22.10)
Exit Mach Number	0.110
Inlet Hub/Tip Ratio	0.600
Rotor Speed, rpm	22,182
Exit Blade Height, cm (in)	0.76 (0.30)

In the three spool, 266,880 N (60,000 lb) thrust size, the high pressure compressor was an all axial configuration. A centrifugal design was analyzed, but in this flow size 26.8 kg/sec/m^2 (5.5 lb/sec/ft^2), it could not compare with an all axial compressor on a component performance basis.

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Because of the emphasis on reduced length, the intermediate and high pressure compressors in the all axial compressor were close-coupled, resulting in the addition of 1 stage. Yet, this still yielded a shorter compression system than if the high pressure compressor had been designed to a high inlet specific flow and had required a transition duct between itself and the intermediate pressure compressor. Table 4.2-VII compares the high pressure compressor candidates to the reference engine.

TABLE 4.2-VII
SUMMARY OF FLOWPATH CANDIDATE HIGH PRESSURE COMPRESSORS

		Candidate							
		1	2	3	4	5	6	7	8
	Reference Engine	1.5 FPR 2 Spool	3 Spool (No LPC)	3 Spool (With LPC)	1.3 FPR 2 Spool	2 Spool 46 OPR	2 Spool Axi-Cent.	2 Spl DD 1.5 FPR	2 Spl DD 1.7 FPR
Pressure Ratio	14.0	20.0	8.49	5.00	20.00	20.00	6.0x3.35	20.0	20.0
Inlet Corrected Flow,	35.2	32.0	15.39	9.82	32.0	15.922	15.9	32.0	32.0
kg/sec (lb/sec) $W\sqrt{\theta/\delta}$	(77.7)	(70.7)	(33.95)	(21.65)	(70.7)	(35.104)	(35.1)	(70.7)	(70.7)
Corrected Tip Speed, m/sec	379	440	436	363	440	440	436	439	439
(ft/sec)	(1244)	(1445)	(1431)	(1193)	(1445)	(1445)	(1432)	(1443)	(1443)
Number of Stages	10	11	7	7	11	11	6	11	11
Number of Airfoils	1265	1014	1059	837	1014	1014	537	1014	1014
Inlet Hub/Tip Ratio	0.560	0.490	0.708	0.760	0.490	0.490	0.604	0.489	0.489
Exit Hub/Tip Ratio	0.923	0.890	0.906	0.890	0.890	0.890	0.840	0.890	0.890
Average Aspect Ratio	1.52	1.50	1.50	1.50	1.50	1.50	1.92	1.50	1.50
Average Gap/Chord	0.892	0.967	0.910	0.983	0.967	0.967	0.985	0.967	0.967
Flow Coefficient	0.560	0.624	0.430	0.385	0.626	0.620	0.599	0.624	0.624
Flow/Unit Area, $lbm/sec/ft^2$	38.0	39.9	31.0	29.3	39.9	39.9	39.98	39.98	39.98
($kg/sec/m^2$)	(185.4)	(194.7)	(151.3)	(143.0)	(194.7)	(194.7)	(195.1)	(195.1)	(195.1)
Exit Mach Number	0.291	0.250	0.270	0.250	0.250	0.250	0.450	0.249	0.249
Reaction, Average	0.50	0.682	0.700	0.700	0.682	0.682	0.656	0.682	0.682
Rotor Speed, rpm	13,176	17,640	20,490	20,710	17,640	23,833	22,182	17,638	17,638
Average Diffusion Factor	0.456	0.462	0.458	0.463	0.463	0.463	0.446	0.461	0.461
Average Airfoil Row Loading	0.413	0.381	0.398	0.403	0.381	0.381	0.352	0.382	0.382

4.2.2.6 Diffuser/Combustor

The axial compressor system incorporated a low loss diffuser, and a high dome flow, high mixing rate combustor as illustrated in Figure 4.2-12. It offered the potential for shorter length and better performance relative to the combustion system of the reference engine. The diffusion system consisted of multiple channel tubes to direct and diffuse compressor discharge air to various regions of the combustor. The channel tubes, which capture the relatively cooler central core of compressor discharge air, direct the air to the inner and outer cavities surrounding the combustor. This air cools the combustor liner, turbine rotors and turbine airfoils. Hotter boundary layer compressor discharge air is fed directly to the combustor dome to act as primary combustion air.

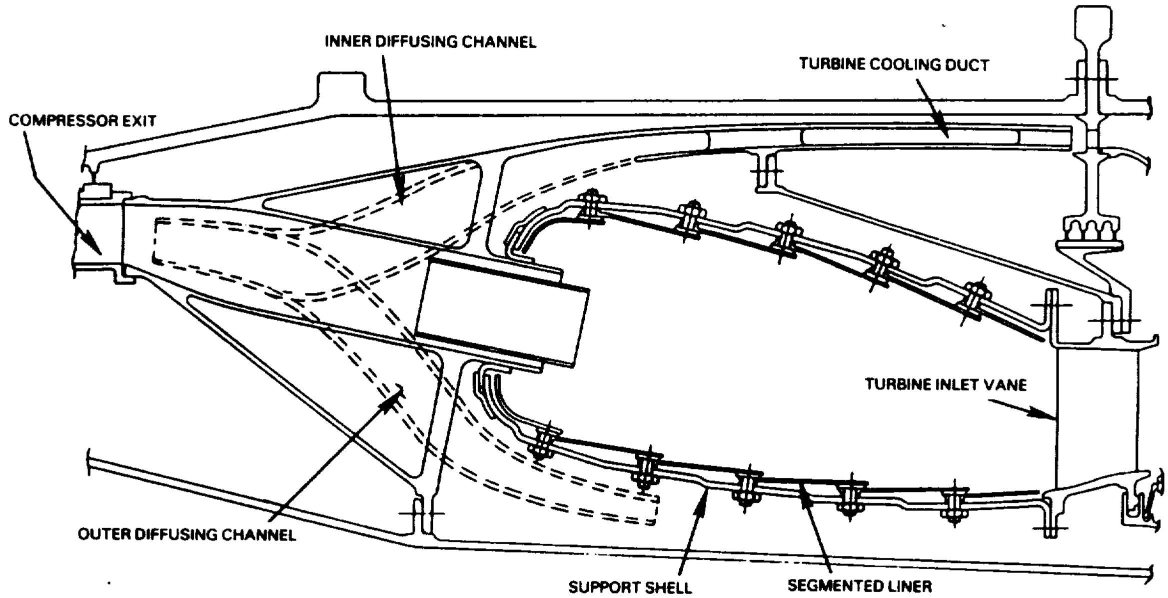

Figure 4.2-12 Advanced Technology Axial Flow Diffuser/Combustor

The centrifugal compressor system used a radial in-flow combustor with a series of conventional tangential pipe diffusers that direct the air into the cavity surrounding the combustor. This configuration is illustrated in Figure 4.2-13.

Both combustors incorporated an advanced segmented liner attached to a support shell. The liners provided the required durability, with acceptable air flow at the very high pressure and temperature environment of the engine.

4.2.2.7 High Pressure Turbine

The two spool candidates, like the reference engine, had two stage high pressure turbines. Compared to the reference engine, the turbine cooling air level for the two stage configurations was reduced, blade and vane trailing edge thicknesses were reduced, and AN^2 was increased to reflect the advanced time frame. The high pressure turbines of the three spool candidates required the same technology but were single stage configurations. Table 4.2-VIII compares the high pressure turbine candidates to the reference engine.

4.2.2.8 Intermediate Pressure Turbine

As indicated in Table 4.2-IX, the intermediate pressure turbine of candidate 2 had a design velocity ratio of 0.485. The pressure split of 1.3 x 8.2 x 6.0 caused the intermediate pressure turbine to have a relatively high work requirement with slow high pressure rotor speed (determined by the 441 m/sec (1450 ft/sec) tip speed criteria of the intermediate compressor).

The intermediate pressure turbine of candidate 3, on the other hand, had a lower work requirement and higher rpm. This resulted in the intermediate pressure turbine of candidate 3 having inherently better design parameters (velocity ratio and AN^2).

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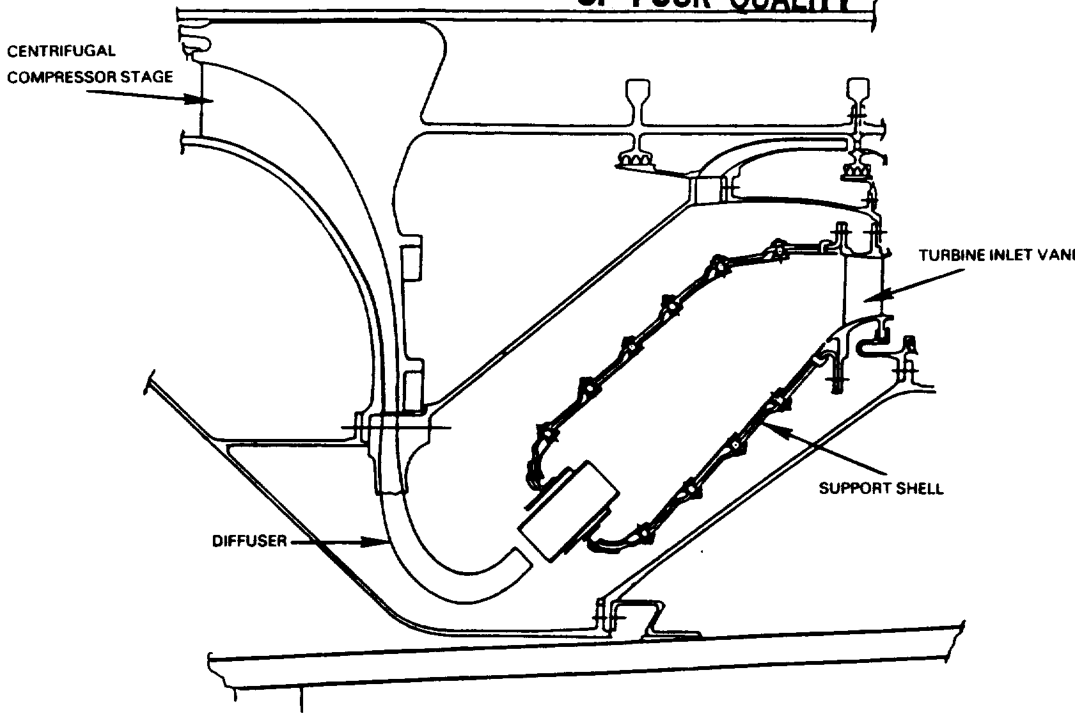

Figure 4.2-13 Advanced Technology Centrifugal Flow Diffuser/Combustor

TABLE 4.2-VIII
SUMMARY OF FLOWPATH CANDIDATE HIGH PRESSURE TURBINE DESIGNS

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR	6 2 Spool Axi-Cent.	7 2 Spl DD 1.5 FPR	8 2 Spl DD 1.7 FPR
Expansion Ratio	4.00	4.60	3.22	2.50	4.80	4.30	4.78	4.593	4.593
Number of Stages	2	2	1	1	2	2	2	2	2
Mean Velocity Ratio	0.64	0.65	0.59	0.63	0.65	0.65	0.65	0.65	0.65
AN^2 Redline $(X10^{10})$	5.0	6.0	6.0	6.2	6.0	6.0	6.0	6.0	6.0
Rim Speed at Redline, m/sec	397	445	525	498	445	426	440	445	445
(ft/sec)	(1305)	(1460)	(1724)	(1636)	(1460)	(1400)	(1446)	(1461)	(1461)
Max. Blade Turning, degrees	98	96	102.3	92.7	97.0	105.6	102.8	96.0	96.0
Flow Coefficient	0.350	0.345	0.337	0.290	0.347	0.301	0.275	0.345	0.345
Exit Swirl Angle, degrees	15.0	14.3	37.1	29.5	14.5	15	19.6	24.5	24.5
Exit Axial Mach Number	0.310	0.350	0.351	0.261	0.362	0.303	0.280	0.350	0.350
Number of Airfoils	149	130	57	49	129	101	104	130	130
Mean Diameter, cm	61.7	50.8	50.2	47.7	51.0	36.3	40.3	50.8	50.8
(in)	(24.3)	(20.0)	(19.8)	(18.8)	(20.1)	(14.30)	(15.9)	(20.0)	(20.0)
Trailing Edge Thickness, cm	0.165	0.101	0.101	0.101	0.101	0.101	0.101	0.101	0.101
(in)	(0.065)	(0.040)	(0.040)	(0.040)	(0.040)	(0.040)	(0.040)	(0.040)	(0.040)
Total Cooling Air (HPT + LPT), %	18.2	8.0	10.0	10.0	10.0	11.1	11.4	10.0	10.0

TABLE 4.2-IX
SUMMARY OF FLOWPATH CANDIDATE INTERMEDIATE PRESSURE
TURBINE DESIGNS

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR
Expansion Ratio	N/A	N/A	1.91	1.935	N/A	N/A
Number of Stages			1	1		
Mean Velocity Ratio			0.485	0.630		
AN^2 Redline			2.90	5.0		
Rim Speed at Redline, m/sec			273	373		
(ft/sec)			(896)	(1227)		
Max. Blade Turning, degrees			109.5	85.8		
Flow Coefficient			0.668	0.478		
Exit Swirl Angle, degrees			36.7	11.1		
Exit Axial Mach Number			0.361	0.340		
Number of Airfoils			69	65		
Mean Diameter, cm			52.5 cm	49.27		
(in)			(20.7)	(19.40)		
Trailing Edge Thickness, cm			0.101	0.101		
(in)			(0.040)	(0.040)		

4.2.2.9 Turbine Transition Section

Typically, high bypass ratio/low fan pressure ratio flowpaths have long turbine transition sections. This is demonstrated in the reference engine and both of the direct drive configurations (candidates 7 and 8) where the combination of high bypass ratio/low fan pressure ratio resulted in slow low pressure rotor speed. Also, the high pressure ratio of the low pressure spool in the two direct drive candidates resulted in high work requirements for the low pressure turbine. This translates to large radial offset between the high and low pressure turbines.

However, in the geared fan designs, low pressure turbine speed was selected independently of fan performance considerations. For example, the work requirement of the low pressure turbine of candidate 1 was 35 to 40 percent higher than that of the reference engine. However, the low pressure rotor speed of candidate 1 scaled to the same size as the reference engine was 1.5 times greater. This resulted in significantly less radial offset between the high and low pressure turbines.

The transition lengths in all of the applicable candidate flowpaths were set by a mean angle criterion of 30 degrees (i.e., measured from the mean exit elevation of the high pressure turbine to the mean inlet elevation of the low pressure turbine). Table 4.2-X compares the turbine transition sections of the candidate flowpaths to the reference engine.

TABLE 4.2-X
SUMMARY OF FLOWPATH CANDIDATE TURBINE TRANSITION SECTION DESIGNS
(High-to-Low Pressure Turbine)

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR	6 2 Spool Axi-Cent.	7 2 Spl DD 1.5 FPR	8 2 Spl DD 1.7 FPR
Length, cm (in)	23.36	9.95	Close-	Close-	12.75	4.44	3.04	23.87	13.56
	(9.20)	(3.92)	Coupled	Coupled	(5.02)	(1.75)	(1.20)	(9.40)	(5.34)
Mean Angle, degrees	25.0	30.0	N/A	N/A	30.0	17.0	----	30.0	30.0
Area Ratio	1.22	1.10			1.10	1.10	----	1.10	1.10
Equivalent Conical Angle	3.80	3.43			2.60	6.25	----	1.25	2.42
Inner Diameter Radius, cm	11.43	6.09			7.87	1.27	Close-	14.73	8.35
(in)	(4.50)	(2.40)			(3.10)	(0.50)	Coupled	(5.80)	(3.29)

4.2.2.10 Low Pressure Turbine

The low pressure turbine work requirements for the candidate flowpaths were higher than that of the reference engine (higher bypass ratio/lower fan pressure ratio and increased supercharging in some instances). However, the increased rotor speed (associated with geared fan designs) of the low pressure turbine provided additional work capability without a significant diameter increase.

In the reference engine, the velocity ratio was limited and in the direct drive flowpath candidates, both the velocity ratio was limited and the number of stages increased to limit the radial offset between the high and low pressure turbines. Table 4.2-XI compares the low pressure turbines of the candidate flowpaths to the reference engine.

TABLE 4.2-XI
SUMMARY OF FLOWPATH CANDIDATE LOW PRESSURE TURBINE DESIGN

	Reference Engine	Candidate 1 1.5 FPR 2 Spool	2 3 Spool (No LPC)	3 3 Spool (With LPC)	4 1.3 FPR 2 Spool	5 2 Spool 46 OPR	6 2 Spool Axi-Cent.	7 2 Spl DD 1.5 FPR	8 2 Spl DD 1.7 FPR
Expansion Ratio	6.10	10.8	7.73	9.705	11.79	7.932	8.33	9.85	9.21
Rotor Speed, rpm	3620	7245	7411	7377	6891	11,870	11,183	2980	3390
Number of Stages	5	5	5	5	5	4	4	7	7
Configuration	Offset	Offset	Close-Coupled	Close-Coupled	Offset	Offset	Close-Coupled	Offset	Offset
AN^2 at Redline ($X10^{10}$)	1.68	6.60	5.74	6.85	6.60	6.60	6.62	1.15	1.35
Mean Velocity Ratio	0.49	0.60	0.622	0.564	0.600	0.600	0.58	0.394	0.394
Max. Blade Turning, degrees	114	97.5	90.0	105.7	99.1	98.0	103.3	125	121
Exit Swirl Angle, degrees	25	7.5	11.5	10.0	8.0	14.0	27.4	35.4	33.8
Exit Axial Mach Number	0.38	0.45	0.424	0.43	0.450	0.450	0.450	0.450	0.446
Flow Coefficient	0.72	0.58	0.65	0.64	0.565	0.566	0.556	1.000	1.08
Exit Tip Diameter, cm (in)	132.0	106.4	105.1	104.6	112.2	67.5	70.3	126.39	115.29
	(52.0)	(41.9)	(41.4)	(41.2)	(44.2)	(26.6)	(27.7)	(49.76)	(45.39)
Axial Length, cm	43.6	41.6	42.9	43.1	42.9	22.8	23.92	60.55	56.84
(in)	(17.2)	(16.4)	(16.9)	(17.0)	(16.9)	(9.0)	(9.42)	(23.84)	(22.38)
Clearance Average, mils	20	10	10	10	10	10	10	10	10
Number of Airfoils	1119	812	802	752	796	663	617	1714	1754

4.2.3 Selection of Candidate Flowpaths

In evaluating advanced airplane thrust requirements, it was determined that a potential market existed for both small, short range aircraft in the 111,205 N (25,000 lb) thrust range and large, long range aircraft in the 266,892 N (60,000 lb) thrust range. In addition, both two and three spool configurations are viewed as design options in the future. Finally, in the smaller thrust range, both all axial and axial-centrifugal designs are considered to be acceptable. However, due to critical speed and stiffness considerations, the axial-centrifugal arrangement was considered more viable.

Installed thrust specific fuel consumption trends over the range of cycles evaluated for propulsive efficiency improvement are shown in Figure 4.2-14. The tendency toward high bypass ratio, geared drive, separate exhaust flow is evident with current technology nacelle design, and is stronger with advanced nacelles. Based on these considerations, three flowpaths were selected for further study: candidate 1, candidate 3, and candidate 6. Table 4.2-XII presents a general summary of the above three candidates compared to the reference engine.

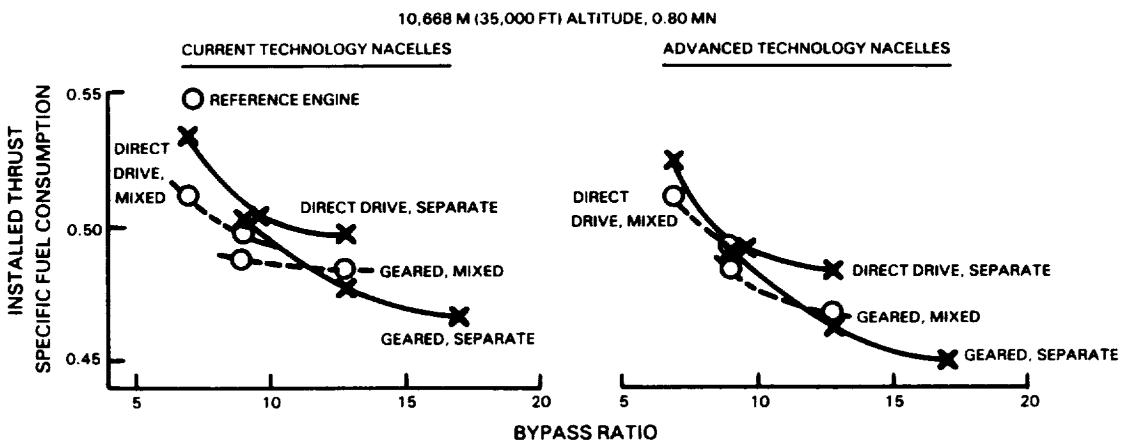

Figure 4.2-14 Thrust Specific Fuel Consumption Comparison of Exhaust and Drive Types for Current and Advanced Nacelle Technology

TABLE 4.2-XII
SUMMARY OF FINAL THREE FLOWPATH CANDIDATES

	Reference Engine	Candidate 1: 1.5 FPR 2 Spool	Candidate 3: 3 Spool With LPC	Candidate 6: 2 Spool Axi-Cent.
CYCLE				
W_{AT2}, kg/sec (lb/sec)	679 (1498)	1184 (2612)	1184 (2612)	504 (1112)
Fan Pressure Ratio	1.65	1.53	1.53	1.53
Bypass Ratio	7.2	12.8	12.8	12.5
Overall Pressure Ratio	38.6	64.0	64.0	55.0
Combustor Exit Temp., °C (°F)	1268 (2315)	1329 (2425)	1329 (2425)	1329 (2425)
FAN				
OD Fan Pressure Ratio	1.65	1.50	1.50	1.50
Tip Diameter, cm (in)	215.9(85.0)	271.2(106.8)	274.0(107.9)	172.2(67.8)
Inlet Hub/Tip Ratio	0.340	0.260	0.260	0.260
Corrected Tip Speed, m/sec (ft/sec)	441 (1450)	356 (1170)	356 (1170)	356 (1170)
Number of Airfoils	36	24	24	24
LOW PRESSURE COMPRESSOR				
Pressure Ratio	1.84	2.52	2.05	2.15
Number of Stages	4	3	3	3
Average Aspect Ratio	2.30	1.90	1.90	1.91
Average Gap/Chord Ratio	0.930	1.00	1.00	0.992
Rotor Speed, rpm	3620	7245	7377	11,183
Number of Airfoils	764	253	186	224
INTERMEDIATE PRESSURE COMPRESSOR				
Pressure Ratio	N/A	N/A	4.92	N/A
Number of Stages			5	
Average Aspect Ratio			1.50	
Average Gap/Chord Ratio			1.10	
Rotor Speed, rpm			10,856	
Number of Airfoils			346	
INTERMEDIATE CASE				
Length, cm (in)	39.6(15.6)	37.5(14.8)	23.62(9.30)	18.54(7.30)
ID Radius, cm (in)	24.89(9.80)	21.08(8.30)	(IPC-HPC) 8.20(3.23)	7.62(3.00)
HIGH PRESSURE COMPRESSOR (Axial Only)				
Pressure Ratio	14.0	20.0	5.00	6.00
Number of Stages	10	11	7	6
Rotor Speed, rpm	13,176	17,640	20,710	22,182
Number of Airfoils	1265	1014	837	537
HIGH PRESSURE COMPRESSOR (Centrifugal)				
Pressure Ratio	N/A	N/A	N/A	3.35
Specific Speed				72.5
Maximum Tip Speed, m/sec (ft/sec)				651 (2139)

TABLE 4.2-XII (continued)

	Reference Engine	Candidate 1: 1.5 FPR 2 Spool	Candidate 3: 3 Spool With LPC	Candidate 6: 2 Spool Axi-Cent.
COMBUSTOR				
Configuration	Axial	Axial	Axial	Radial Inflow
Length, cm (in)	38.1 (95.0)	35.0 (13.8)	35.0 (13.8)	27.4(10.8)
Space Heating Rate, M Btu/hr (ft^3) (atmos)	5.1	7.0	7.0	3.0
Combustion Length, cm (in)	20.5 (8.1)	17.7 (7.0)	17.7 (7.0)	17.7 (7.0)
HIGH PRESSURE TURBINE				
Expansion Ratio	4.00	4.60	2.50	4.78
Velocity Ratio	0.64	0.65	0.63	0.650
Number of Stages	2	2	1	2
Number of Airfoils	149	130	49	104
AN^2 ($\times 10^{10}$), (in^2)(rpm^2)	5.0	6.0	6.2	6.0
INTERMEDIATE PRESSURE TURBINE				
Expansion Ratio	N/A	N/A	1.94	N/A
Velocity Ratio			0.630	
Number of Stages			1	
Number of Airfoils			65	
AN^2 ($\times 10^{10}$), (in^2)(rpm^2)			5.0	
TRANSITION DUCT				
Length, cm (in)	23.36(9.20)	9.95(3.92)	Close-Coupled	3.04(1.20)
Area Ratio	1.22	1.10		----
LOW PRESSURE TURBINE				
Expansion Ratio	6.10	10.8	9.705	8.33
Velocity Ratio	0.49	0.60	0.564	0.580
Number of Stages	5	5	5	4
Number of Airfoils	1119	812	752	617
Max. Tip Diameter, cm (in)	132.0(52.0)	106.4(41.9)	104.6(41.2)	70.3(27.7)
AN^2 ($\times 10^{10}$), (in^2)(rpm^2)	1.68	6.60	6.85	6.62

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4.3 MECHANICAL FEASIBILITY STUDIES

The candidate flowpaths provided the physical envelope for the fan, compressor, and turbine sections of the possible future turbofan engines. The mechanical feasibility studies used these envelopes to establish representative propulsion system cross sectional layouts. In addition, major emphasis was placed on assessing the most critical elements of the propulsion system concept.

4.3.1 Candidate Flowpath Support System Definition

The components of the three selected candidate flowpaths were characterized by very high speed, small diameter compressors and turbines, and a relatively large diameter, slow speed fan driven through gears by the high speed low pressure turbine.

The two spool 266,892 N (60,000 lb) thrust engine, candidate 1, evolved into the simplest mechanical arrangement utilizing three major support cases as shown in Figure 4.3-1. The low pressure compressor inlet case and the intermediate case were tied together by an outer shell, providing the central hub for the forward section of the engine. This structure provides support for the fan, its shaft and bearings, the fan reduction gearbox, and the forward bearings of the low and high speed shafts. Additionally, the intermediate case provides a base for the front mount plane, compressor case structure, and the inner shell of the nacelle which will support the high pressure spool and low pressure turbine cases.

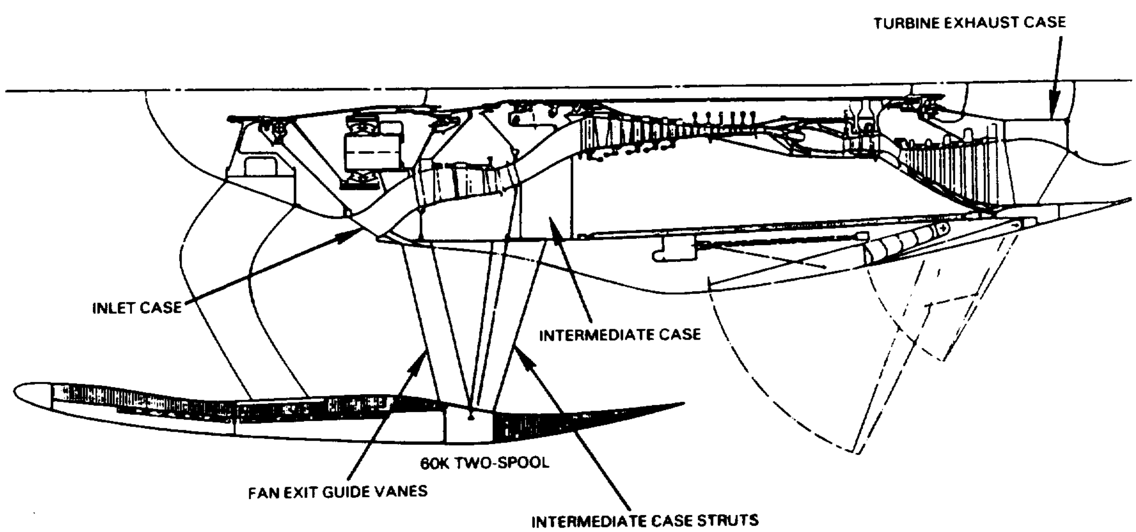

Figure 4.3-1 Cross Section of the Two Spool 266,892 N (60,000 lb) Thrust Engine, Candidate 1, Showing Three Major Support Cases

The 266,892 N (60,000 lb) thrust three spool configuration, candidate 3, and the 111,205 N (25,000 lb) thrust two spool configuration, candidate 6, require additional support cases because of added bearing location requirements. Candidate 3, shown in Figure 4.3-2, requires two additional frames to support bearings at the front end of the high pressure spool and between the intermediate and low pressure turbines. The mid-compressor frame for candidate 6, shown in Figure 4.3-3, is required to maintain close axial spacing between the centrifugal compressor impeller and its casing.

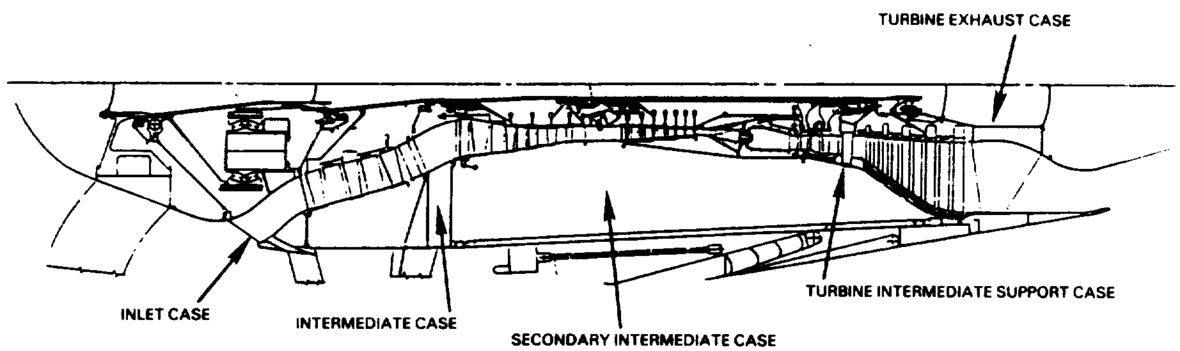

Figure 4.3-2 Cross Section of the Three Spool 266,892 N (60,000 lb) Thrust Engine, Candidate 3, Showing Additional Support Cases

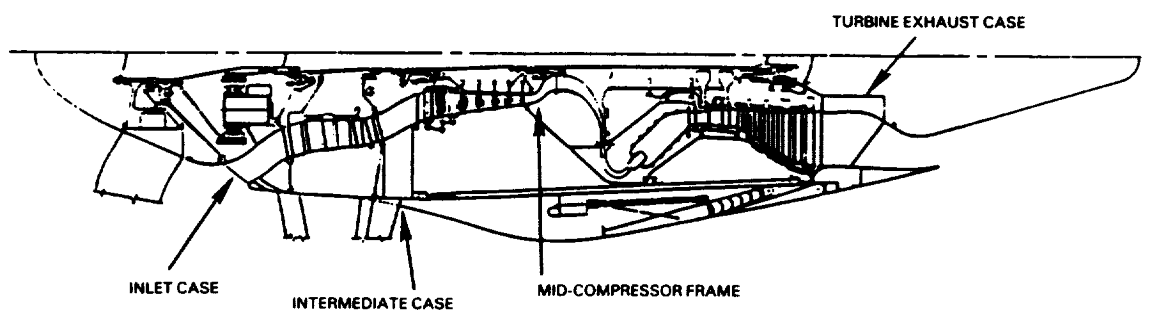

Figure 4.3-3 Cross Section of the Two Spool 111,205 N (25,000 lb) Thrust Engine, Candidate 6, Showing Mid-Compressor Frame

The large diameter fan is driven by the high speed low pressure turbine through a 3:1 reduction gearbox. The high horsepower (approximately 55,000 for the 60K configuration and approximately 23,000 for the 111,205 N (25,0001b) configuration) transmission presents challenges in obtaining a flightweight highly efficient configuration.

The locations of the rotor bearing supports were established in part by modules formed by the individual components, rotor critical speed requirements, and the environment surrounding possible bearing compartment locations. A piggyback bearing arrangement supporting the turbine sections would avoid the very high pressure and temperature environment occurring beneath the combustor region of the engine. Bearings themselves will be beyond current DN levels, requiring advancements in bearing design and bearing lubrication technology.

4.3.2 Feasibility of Critical Elements

Several critical elements related to the mechanical design of the three configurations were assessed for feasibility:

- o reduction gear,
- o high pressure rotor,
- o main bearing speeds,
- o advanced nacelle/reverser.

4.3.2.1 Reduction Gear

The cycle studies resulted in selection of a large diameter, slow speed fan, driven through a reduction gear by a high speed turbine section. The objective of this analysis was to determine the feasibility of a long life reduction gear capable of efficiently transmitting the high horsepower required. For this configuration, a reduction gear would have to be developed that was reliable, highly efficient and light.

To meet the objective, material properties were projected, the gearset and bearings were sized and efficiency levels were determined. In addition, overall design goals were established. These included a mean time between repair of greater than 15,000 hours and a cruise efficiency of 99.3 percent. The design philosophy included stiff shafting and casing to minimize deflections and deformations, and long life gears and gearbox bearings.

The projected level of technology available for reduction gears by the year 2005 is compared to reference engine technology in Table 4.3-I.

TABLE 4.3-I
TECHNOLOGY AVAILABLE FOR REDUCTION GEAR CONCEPTS

	Reference Engine	Technology Projected for 2005
Gears		
Materials	AMS 6265	Rapid Solidification Rate Powder Product Advanced Metallurgy
Bending Fatigue Limit		
Unidirectional, MPa(1)	344.7	482.6
(psi(1))	(50,000)	(70,000)
Reversed Bending, MPa(1)	282.7	406.8
(psi(1))	(41,000	(59,000)
Hertz Stress Limit, MPa(1)	868.7	1,241.1
(psi(1))	(126,000	(180,000)
Pitch Line Velocity Limit, m/min	9,144	12,192
(ft/min)	(30,000)	(40,000)
Bearings		
Materials	CVM M50	Advanced Metallurgy
System Design Life Requirement, hr	18,000	18,000
Material/Lubrication Life Factor	6 to 12	60 to 90

TABLE 4.3-I (Continued)

	Reference Engine	Technology Projected for 2005
Housings		
Materials	Aluminum, Magnesium	Composites (Metal Matrix, etc.)
Lubricant		
Fluids	Mil 23699 Type II	Synthesized Hydrocarbon Fluid
Oil Inlet Temp, °C (°F)	65 (150)	121 (250)
Allowable Temperature Rise, °C (°F)	4 to 10 (40) to (50)	37 to 48 (100) to (120)
Load Carrying Ability, kg/cm (lb/in)	2460 to 4305 (2000 to 3500)	6150 to 7995 (5000 to 6500)
Flash Temperature Index, °C (°F)	135 (276)	204 (400)

(1)Typical Gear Allowable Stress - 3 Sigma With a Coefficient of Variation = 0.1, 10^{10} Cycles

With aggressive development programs, the strength of gear materials is anticipated to improve 40 to 45 percent, allowing smaller gears than in service today. Programs for bearing materials and material/lubrication improvements would allow gear (pinion) bearing proportions to be in line with the gears. The potential improvements in lubricants would be needed for load carrying capabilities and operating temperatures. These increased temperature capacities would also allow a reduction in heat exchanger volume and weight. Housing materials and construction will probably evolve through high strength castings toward composites such as metal matrices, resulting in a lightweight, stiff structure.

A potential reduction gear, using the projected technologies is shown in Figure 4.3-4. The gear set is a planetary system with input power from the power turbines to the sun gear transmitted through five fixed star gears to the rotating ring gear. The overall gear ratio is 3.12:1. The inlet case provides the primary support for the gear system (five star gears) and cone supports fore and aft for the shaft bearings. The fan shaft is supported at the front by a ball thrust bearing and at the rear by an intershaft roller bearing to the input shaft (low pressure compressor front hub) from the low pressure turbine. The inlet case is a major structural case, forming part of the central hub structure of the engine.

Using the potential technology advancements, numerous planetary gear and bearing sets were sized and analyzed for life and power loss characteristics. Ring gear pitch diameters analyzed ranged from 20 to 36 inches while gear diametral pitch ranged from 4 to 8. Diametral pitch is a measure of gear tooth proportions; two times the reciprocal of diametral pitch is approximately the tooth height. In general, a higher (finer) diametral pitch results in increased efficiency and a wider gear.

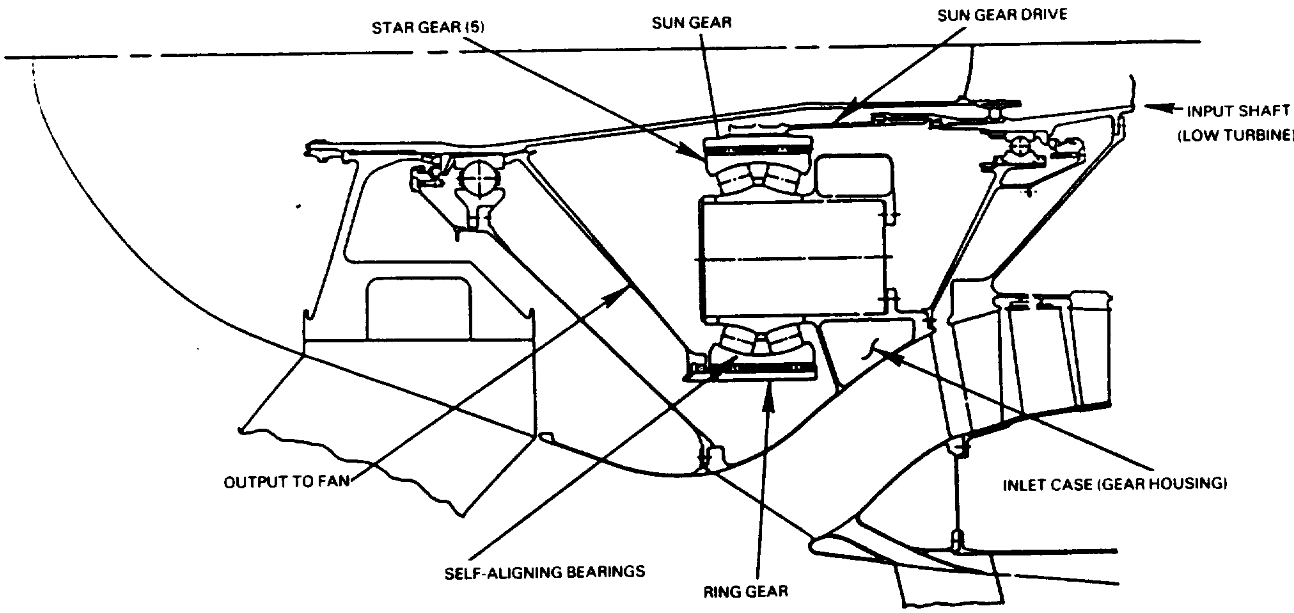

Figure 4.3-4 Fan Speed Reduction Gear

A tabulation of characteristics of the potential reduction gear developed with projected technologies is presented in Table 4.3-II. The efficiency goal of 99.3 percent was attained at sea level takeoff. The general trend is for efficiency to degrade as power levels decrease, but advanced concepts such as regulating oil flow to meet cooling requirements, hence, minimizing churning losses, indicate that high efficiency can be achieved at cruise. As a result of the reduction gear feasibility study, it was concluded that a reduction gear and bearing set could be designed to achieve the desired life and efficiency goals.

TABLE 4.3-II
CHARACTERISTICS OF POTENTIAL FAN REDUCTION GEAR

	Ring Gear	Pinion	Sun Gear
Pitch Dia, cm (in)	83.820 (33.000)	28.785 (11.333)	26.245 (10.333)
No. of Teeth	198	68	62
Min Width, cm (in)	12.57 (4.95)	12.57 (4.95)	12.57 (4.95)
Diametral Pitch, cm (in)	15.240 (6.000)		

	Horsepower Loss
Sun-Pinion Gear Efficiency, 99.6%	220
Pinion-Ring Gear Efficiency, 99.9%	55
Bearing	61
Churning and other	55
Est. Losses at SLTO	391

Results in approximately 99.3% efficiency at sea level takeoff and cruise power settings.

4.3.2.2 High Pressure Rotor

Because the high pressure rotors of the flowpath candidates would utilize turbines with high AN^2 values, the rotors must operate at:

- o significantly higher rim speeds (445 m/sec (1460 ft/sec)) redline)
- o lower hub/tip ratios (inlet to high pressure compressor - 0.489)

This challenges both high pressure compressor and turbine rotors and their attachment regions beyond reference engine capabilities. The objective of the high pressure rotor evaluation was to verify the feasibility of developing high pressure rotor critical areas based on available and projected technologies.

The high speed, slender high pressure spool rotor system of the candidate flowpaths was subjected to a first pass critical speed analysis. Both a conventional straight through combustor system and a shorter folded combustor arrangement were used in the assessment.

The cores used in the analysis are compared in Figure 4.3-5. The large, 266,892 N (60,000 lb) thrust class two spool core, candidate 1, was refitted with the folded combustor in part "a" of the figure. The length between rotor supports was shortened by 8.6 inches. Weight estimates of the compressor and turbine rotors used in the analysis are shown in Figure 4.3-6.

First, second and third mode natural frequencies were calculated for the two rotor systems over a range of spring rates for the front and rear rotor support structures, K_1 and K_2. The results are presented in Figures 4.3-7 and 4.3-8. At spring rates of less than about $12.3X10^5$ kg/cm (10^6 lb/in), the first and second modes were characterized by "bounce" and pitch," respectively, which involve very little rotor bending. Correspondingly, rotor strain energy was very low. As the support system was progressively stiffened, a significant amount of rotor bending occurred in these two modes. Also, strain energy became large enough to require damping to keep lateral motion small and maintain tight blade tip clearances. The third mode exhibited high strain energy at all spring rates and was avoided.

The shorter and stiffer rotor system of the folded combustor raises natural frequencies by an amount equivalent to approximately 5000 rpm at a given spring rate in regions where high strain energies exist. For both rotor arrangements, redline speed is below the third mode natural frequency with acceptable margins although damping may be required. By selecting support spring rates in the typical range of $12.3X10^4$ kg/cm (10^5 to 10^6 lb/in), the first and second mode strain energies are low.

It was concluded that the conventional, straight through combustor would provide sufficient safety margins at important critical speed modes. The shorter, folded combustor system, with its inherently higher section pressure loss, would not be required for dynamic stability.

The ability to meet the challenges of the higher speed operation is largely dependent on anticipated increases in strength and temperature capability of rotor materials. The emphasis in turbine design and materials utilization will be to reduce drastically the weight of the turbine foil and attachment regions.

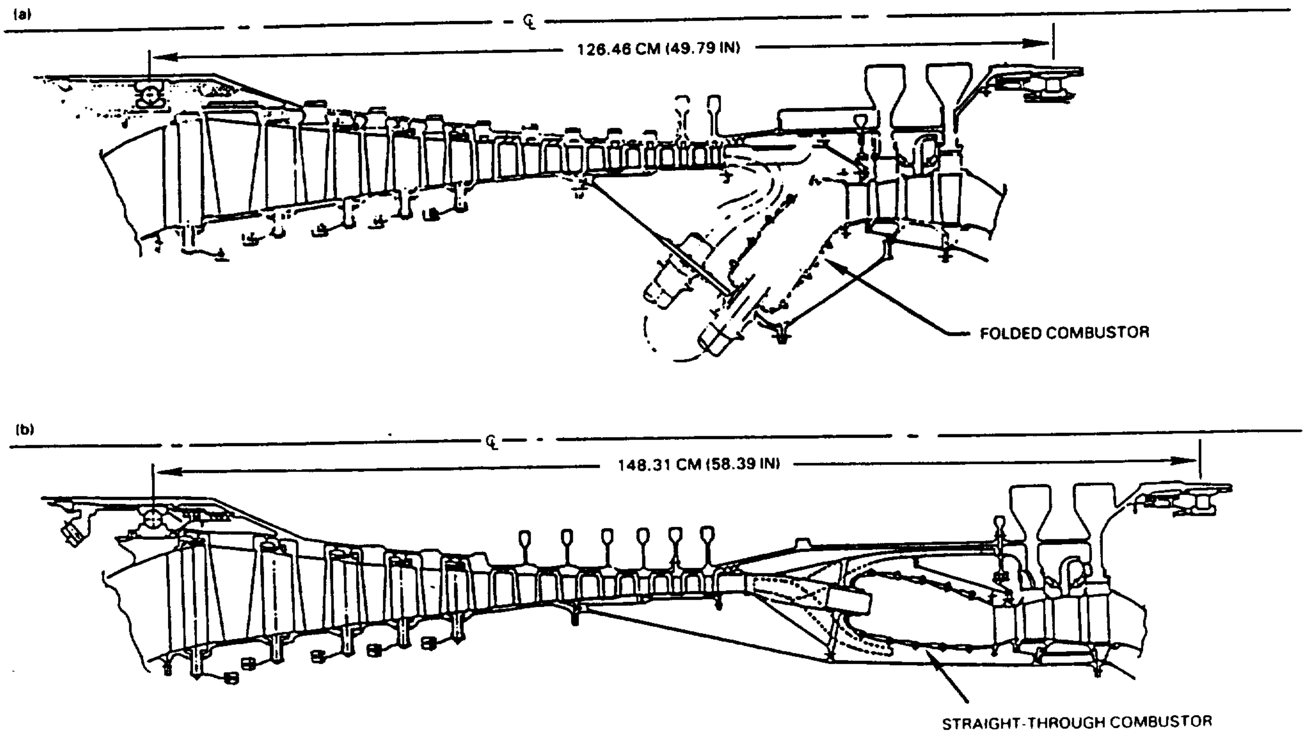

Figure 4.3-5 Cores Used in Critical Speed Analysis of High Pressure Spool Dynamic Stability Examination

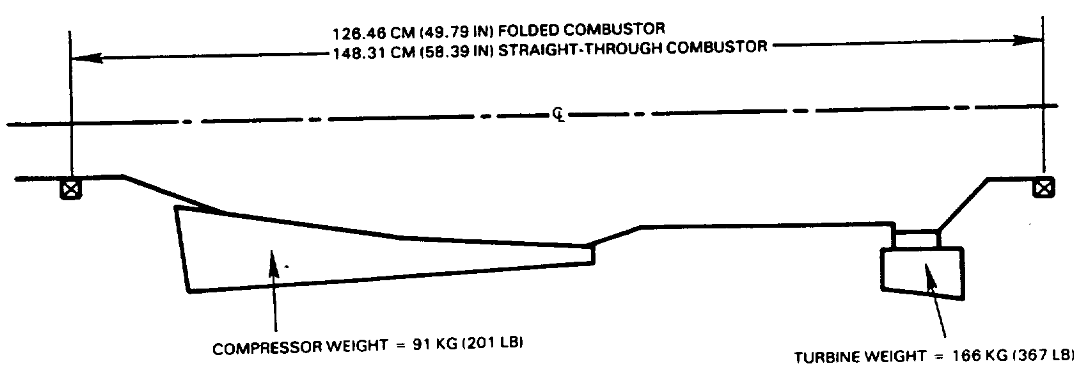

Figure 4.3-6 Weight Estimates of the High Pressure Spools Used in the Critical Speed Analysis

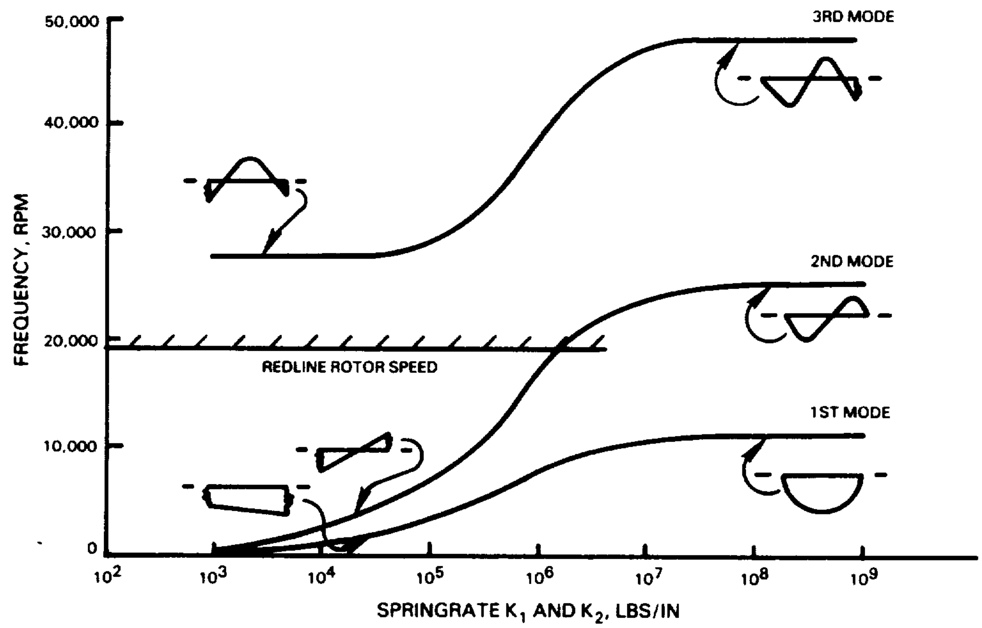

Figure 4.3-7 Results of High Pressure Spool Critical Speed Analysis with the Folded-Back Combustor

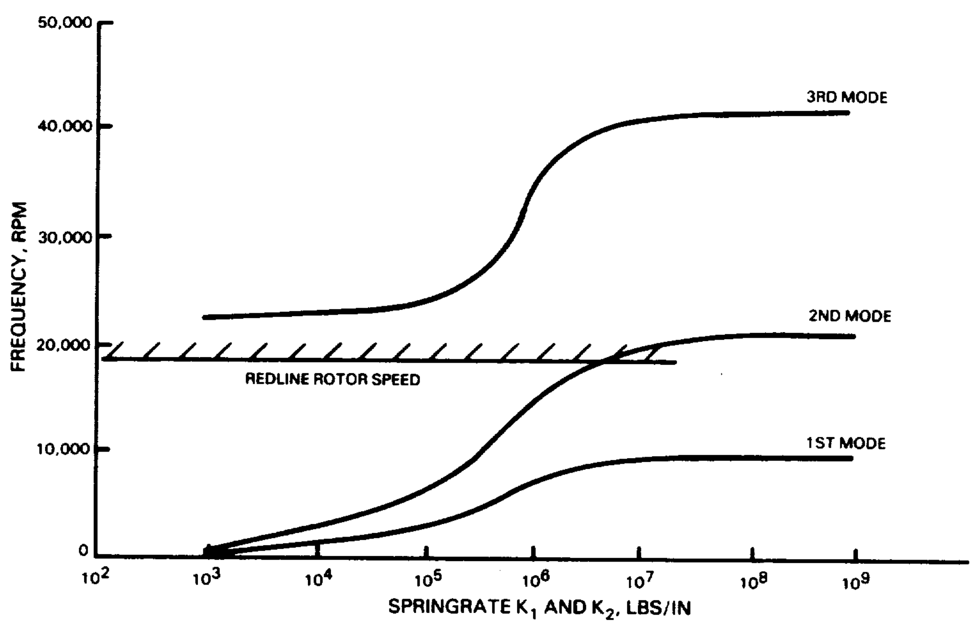

Figure 4.3-8 Results of High Pressure Spool Critical Speed Analysis with the Straight Through Combustor

It is expected that compressor rotors of advanced titanium and nickel alloys in a drum configuration will provide up to a 20 percent increase in strength and durability. Other advancements are expected to include:

- o local reinforcement with polymeric composites to provide a 35 percent weight savings;
- o airfoils fabricated with titanium aluminide alloys that reduce airfoil pulls and increase temperature capability;
- o turbine disks fabricated with new alloys that provide a 25 percent increase in strength and durability;
- o increased rim temperature capabilities to 760°C (1400°F);
- o hybrid disks including bimetal and carbon/carbon reinforcement that provide a 30 percent weight reduction;
- o turbine seals and sideplates that have lower density material, i.e., titanium aluminide and embedded composites, for weight reduction and reduced disk loading;
- o turbine blades with a 204°C (400°F) temperature capability increase.

To meet the objective of the evaluation, the weights and shapes of several disks were determined in the high pressure compressor and turbine based on burst limit criteria. The most critical high pressure compressor disk was the first stage due to the low hub/tip ratio and hub bearing compartment requirements. Attempts at sizing this stage utilizing conventional attachments resulted in a disk with a bore so deep that it occupied area necessary for hub and bearing compartment seals. The disk configuration analyzed, shown in Figure 4.3-9, could use airfoils bonded to the disk rim or integrally machined to the disk.

The high pressure turbine disks were initially sized using attachment and sideplate regions scaled from current turbofan engines. This resulted in the unacceptably heavy and wide turbine disks shown in Figure 4.3-10. To attain acceptable disk shapes, it would be necessary to:

- o reduce sideplate pull by half,
- o eliminate the extended neck portion and dampers of the airfoil,
- o reduce attachment region weight.

It is anticipated that technology development programs will be established for research into these needs and to optimize turbine disks configurations. As a result, it was concluded that rotor disks for both compressor and turbines will be feasible although they will be sensitive to airfoil, attachment, and sideplate weight.

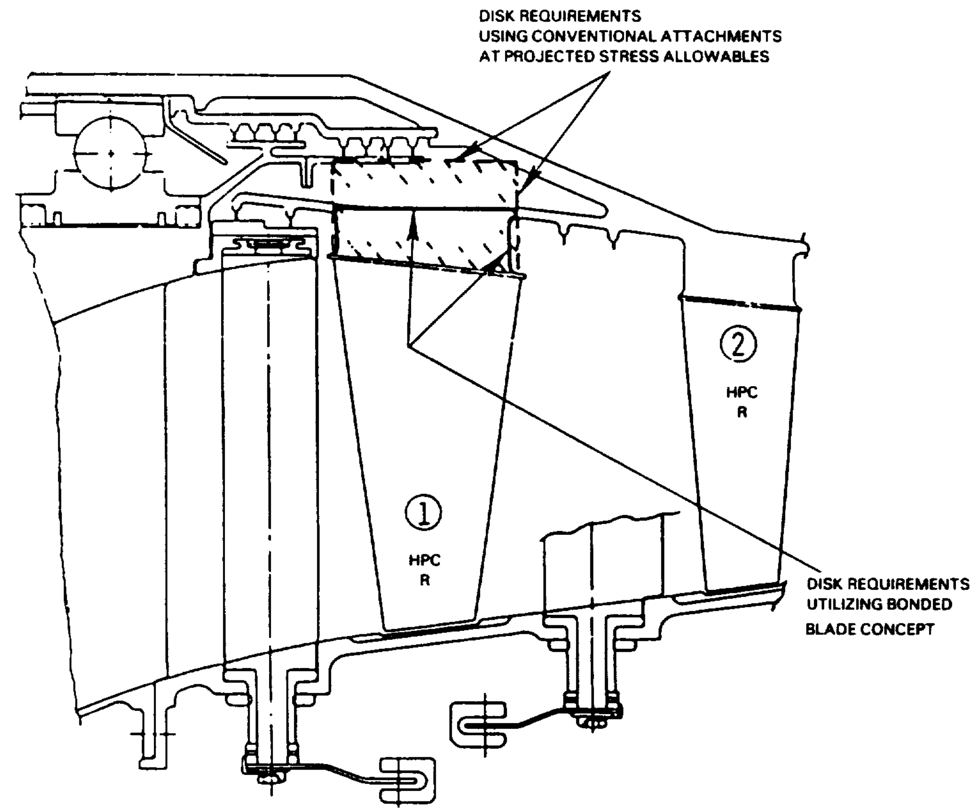

Figure 4.3-9 First Stage High Pressure Compressor Disk Attachment Requirements

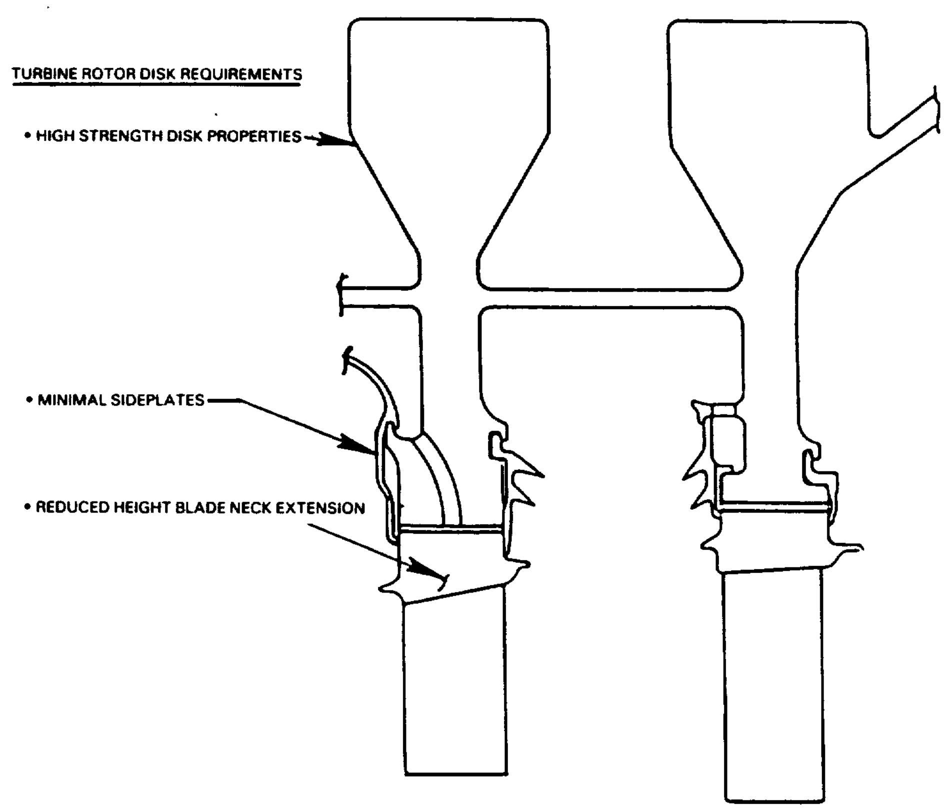

Figure 4.3-10 High Pressure Turbine Disk Feasibility Analysis Configuration

4.3.2.3 Main Bearing Speeds

The advances in turbofan engine technology represented by the candidate flowpaths utilize small high speed core rotors. At these high speeds, increases in centrifugal loading of the rolling elements and race hoop stresses would tend to significantly shorten bearing life. This trend toward higher engine operating speeds would push the bearing life factor, DN, which is the bearing bore in mm multiplied by the shaft speed in RPM, from the current levels of 2.2 million to beyond 3 million as indicated on Figure 4.3-11. Research testing and analysis indicate that these high levels of DN would be attainable through advances in both materials and bearing design technology. This development is essential if the substantial improvements in efficiency and reductions in cost and weight identified for the candidate flowpaths are to be realized.

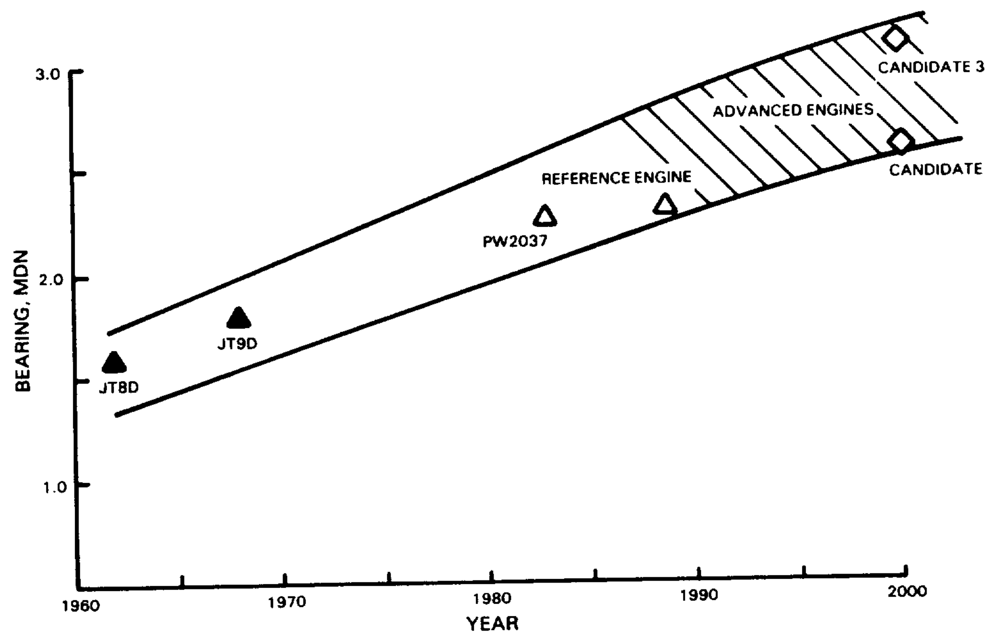

Figure 4.3-11 Historical Trend toward Increased Bearing DN Levels

4.3.2.4 Advanced Nacelle/Reverser

Reference engine nacelle systems applied to the large diameter fans of the candidate flowpaths would result in bulky, heavy nacelles with high drag. A nacelle cowl with a means of stiffening the engine would be mandatory since, structurally, the core, with its small diameter flowpath and casing, is more flexible. Therefore, a study was undertaken to determine the feasibility of providing a low drag nacelle system for high bypass ratio turbofans incorporating load sharing concepts and a suitable reverser system.

To determine the feasibility of such a system, a potential future nacelle/reverser system was developed. The mount arrangement used, illustrated in Figure 4.3-12, was similar to the reference engine configuration. The front mount attached to the rear of the intermediate case and was intended to take thrust, side, vertical, and torque reactions while the rear mount, a simple hanger, reacted only vertical loads.

INTEGRATED ENGINE/NACELLE CONCEPT

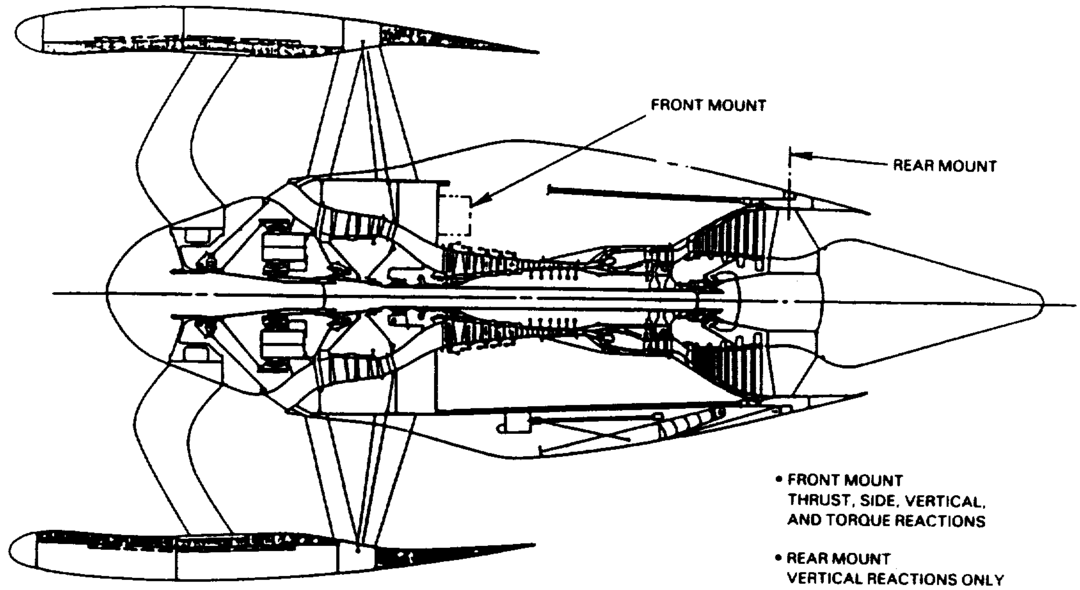

Figure 4.3-12 Potential Future Nacelle/Reverser System Showing Mounting Arrangement

The nucleus of the investigated nacelle/reverser was the hub-spoke-rim system formed by the inlet case, intermediate case and struts, fan exit guide vanes, and structural ring at the nacelle outer cowl. As shown in Figure 4.3-13, the hub elements were formed by the inlet case and intermediate case joined by a structural shell. The hub was very stiff because the inlet and intermediate cases were radially deep and separated axially by a structural shell which enhanced rolling stiffness about an axis perpendicular to the engine centerline. The radial spokes were established by the fan exit guide vanes and intermediate case struts which formed two legs of a triangle and positioned the fan cowl structural ring. The intermediate case struts also had the capability of resisting tangential loads (torque on outer cowl). Fan cowl barrel segments were cantilevered off the structural ring both fore and aft.

The high pressure spool and low pressure turbine extended aft from the inner portion of the intermediate case. Since the diameter of the compressor casing was relatively small, primary support for the core was provided by a load sharing inner nacelle cowl. This provided a bending stiffness approximately 15 times greater than the high pressure spool casing, thus, allowing the engine core to act essentially as a simple support beam between the intermediate case and turbine exhaust case rather than being cantilevered off the intermediate case. This resulted in a rotor-case structure that would provide closer blade-tip clearance potential than the reference engine.

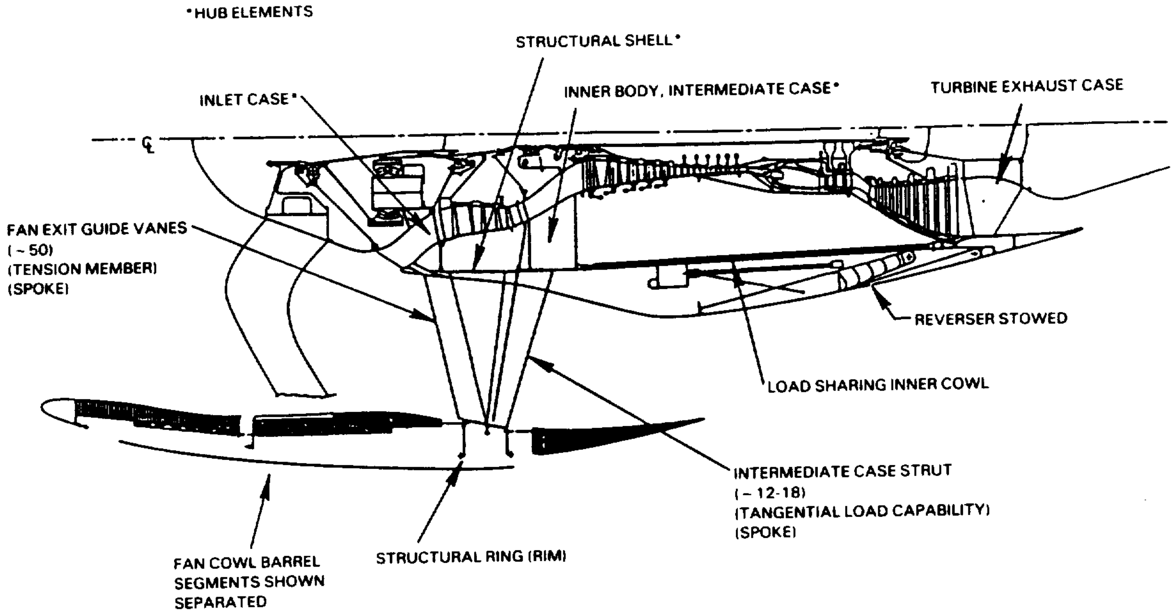

Figure 4.3-13 Potential Future Nacelle/Reverser System Showing Structural Load Path

A review of thrust reverser characteristics of the candidate flowpaths determined that a thrust reverser having an effectiveness of about 1/2 that of current technology reversers would be required. This is illustrated on Figure 4.3-14, which compares a range of reverser effectiveness with that of a JT9D. The projection of a reduced effectiveness reverser led to the following conclusions:

- a simplified reverser could be used,
- the entire reverser might be able to be stowed in the inner cowl,
- the lack of a reverser in the outer cowl would reduce cowl thickness.

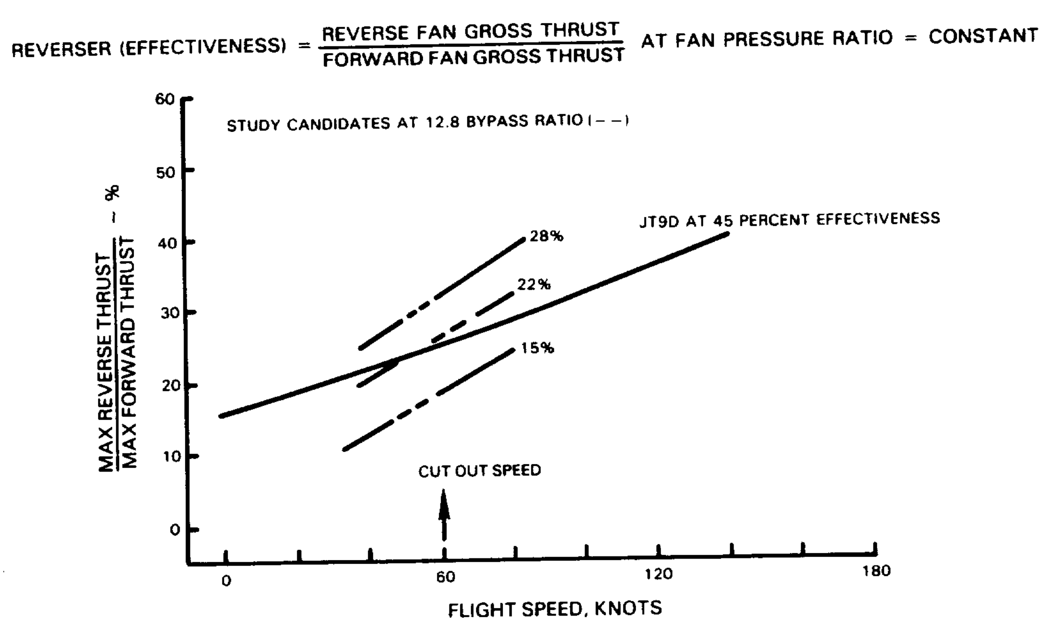

Figure 4.3-14 Comparison of Drag Characteristics of the Candidate Flowpaths with Current Technology Reversers

A potential reverser concept that evolved is shown on Figures 4.3-15 and 4.3-16. It is compared to a conventional reverser in Figure 4.3-17. The potential configuration is an extension of the commonly known "umbrella" reverser concept. Considering the fact that reduced effectiveness would be satisfactory, the maximum diameter of the deployed reverser panels was restricted to less than the cowl outer diameter. Anticipating the possibility that this might result in an ineffective reverser, a secondary flap and cascade were incorporated into the base of the main flaps to bleed stagnant flow and direct it forward, providing reverse thrust.

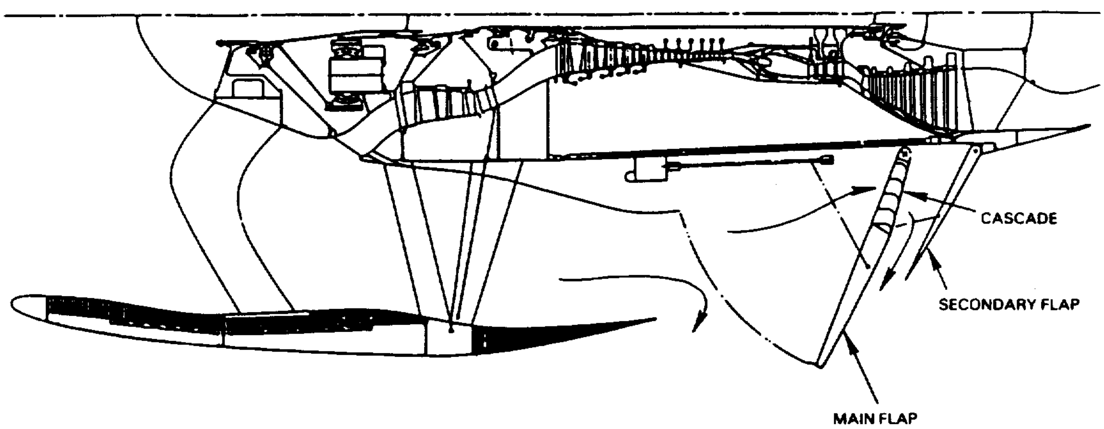

Figure 4.3-15 Potential Future Nacelle/Reverser System Showing Reverser Deployed

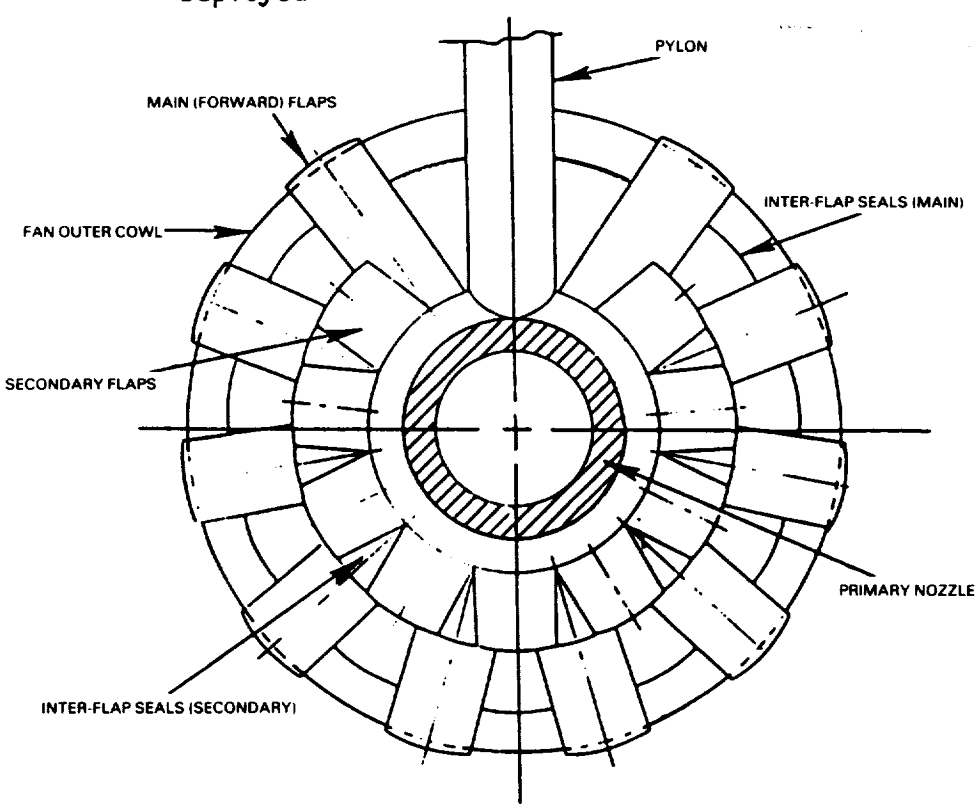

Figure 4.3-16 View of Potential Future Nacelle/Reverser System Looking forward at Deployed Reverser

Sound reduction considerations led to the selection of a plug nozzle for the exhaust case to provide additional surfaces for sound treatment, countering potential turbine noise. In addition, a preliminary parametric study was conducted on the fan region to determine the number of fan exit guide vanes that would be required to effectively cancel aft fan noise. Considerations in the study were:

- o approach (775 rpm), takeoff (2292 rpm);
- o fan exit guide vane range of 15 to 80;
- o up to 5th harmonic of blade passing frequency;
- o inlet and aft radiation;
- o relative annoyance of tones;
- o inlet guide vane interaction tone level.

The results of this study indicated that aft noise radiation would be approximately 2.4 times more annoying than forward radiation. On this basis, the number of exit guide vanes was chosen to be approximately 50.

To achieve the capabilities of such a potential nacelle/reverser concept, technological development programs need to be undertaken to develop:

- o a thin, short, low drag outer cowl;
- o an inner cowl providing structural support and stiffness for the gas generator core;
- o a reverser mechanism housed totally within the inner cowl;
- o advanced sound reduction features.

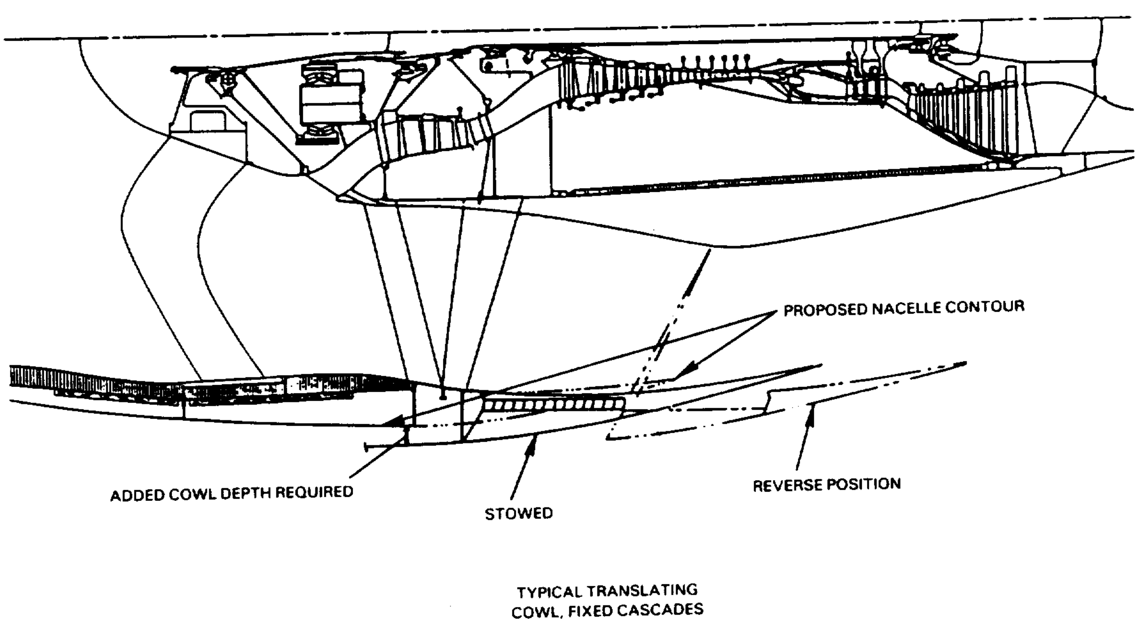

Figure 4.3-17 Comparison of Potential Future Nacelle with Conventional Reverser System

SECTION 5.0
CONCEPT BENEFIT ANALYSIS

The Engine Configuration and Technology Requirement Identification efforts described in Section 4.0 established the component efficiency levels expected in the 2000 to 2010 time frame and identified the technologies needed to meet those efficiency levels. Subsection 5.1 describes the approach used in the benefit analysis, subsection 5.2 summarizes the technologies required and subsection 5.3 describes the assessment of benefits of those technologies.

5.1 BENEFIT ANALYSIS APPROACH

The benefits of the engine technology concepts listed in subsection 4.1.5 were evaluated using three commercial transport airplanes. These airplanes, described in Table 5.1-I, include a small short range twinjet, a medium range trijet and a large long range quadjet.

TABLE 5.1-I
BENEFIT ANALYSIS AIRPLANE DESCRIPTIONS

	Twinjet	Trijet	Quadjet
Design Range, km (n. mi.)	2778 (1500)	5556 (3000)	10186 (5500)
Typical Range, km (n. mi.)	740 (400)	1296.4 (700)	3704 (2000)
Design Passenger Payload	150	440	510
Mach Number at Cruise	0.78	0.80	0.80
Takeoff Gross Weight, N (lb)	667,230 (150,000)	2,224,100 (500,000)	3,291,668 (740,000)
Engine Thrust, N (lb)	111,205 (25,000)	177,928 (40,000)	177,928 (40,000)

Benefits were determined by comparing mission fuel burned and direct operating cost plus "interest" (DOC+I) for these airplanes configured with the advanced technology engine against the same airplanes configured with reference engine technology. The DOC+I groundrules, presented in Table 5.1-II, parallel the 1967 Air Transport Association direct operating cost model with the addition of an interest term to account for cost of money. Except for fuel pricing, it is basically the same as that used in the first phase of the benefit/cost study for the initial screening and ranking.

DOC+I provides a more accurate measure of the worth of engine technology to an airline than DOC. Because it includes the cost of money, it is less sensitive to engine performance changes and more sensitive to engine price changes than DOC. For example, a concept which improved specific fuel consumption by 1 percent would improve DOC by 0.57 percent, but DOC+I would be improved by only 0.44 percent ($0.40/liter ($1.50/gallon) fuel, 150 passenger twinjet). Similarly, assuming the same fuel price and airplane, a concept which would reduce engine price by $100,000 would improve DOC by 0.15 percent while improving DOC+I by 0.29 percent. Thus, DOC+I provides a more conservative measure of the benefit of advanced technologies.

TABLE 5.1-II
DIRECT OPERATING COST PLUS INTEREST ASSUMPTIONS

	1981 Dollars
Fuel Price	$1.00, $1.50, and $2.00
Crew Cost	1981 Boeing Method
Utilization	1981 Boeing
Airframe Maintenance	1981 Boeing
Engine Maintenance	P&WA, Mature Engine
Maintenance Burden	200% on Labor
Airplane/Engine Price	P&WA
Insurance	0.5% Flyaway/Year
Spares	6% Airframe, 30% Engine
Depreciation	Straight Line, 15 Years to 10% Residual
INTEREST	15 PERCENT

5.2 TECHNOLOGY REQUIREMENTS

Increased efficiency in the fan component is expected to result from the shroudless swept fan blade. Removing the part span shroud and sweeping the blade will reduce the penalty associated with high tip speed fans. In addition, high speed reduction gears tied from the low pressure turbine to the fan will allow lower fan pressure ratio and higher bypass ratio than current technology.

Technologies to improve compressor efficiency include higher tip speeds, tighter running clearances and advanced controlled diffusion airfoils. Advanced active clearance control would be required to maintain those tight running clearances. Axial-centrifugal high pressure compressors could provide efficiency improvements for the smaller (111,205 N (25,000 lb) thrust class) turbofan engines. The axial-centrifugal compressor would be less susceptible to bending because it offered a shorter, stiffer high pressure rotor than the longer, more slender all axial compressor for the larger thrust sizes.

Large increases in combustor exit temperature for improved overall efficiency will probably not be required in the future. However, significant thermal efficiency improvements could be achieved by increasing overall pressure ratio with advanced diffuser and combustor materials. Such increases in overall pressure ratio result in an increase in the temperature of the air entering the diffuser/combustor and utilized for cooling turbine vanes and blades. Consequently, the study effort focused on advanced diffusers which could deliver higher quality air with a reduced pressure loss, and on advanced materials to handle the higher temperature air.

Turbine efficiency improvements could result from blades and vanes with reduced trailing edge thickness and small, high speed, increased AN^2 turbines. Blade attachment and materials advances will also be required for future high and low pressure turbines.

Advanced nacelle designs will be required to minimize the external drag penalties of large diameter fans. By reducing the maximum nacelle diameter for a given fan diameter, airplane/engine-installation interaction would be enhanced. In addition, since the high speed core of the future would be more flexible, a nacelle cowl with a means of stiffening the engine would be required.

Many of these potential component improvements will depend on advancements in materials technology. For instance, with aggressive development programs, the strength of gear materials could improve 40 to 45 percent, allowing smaller gears than today's. Development of composites will be necessary for lighter weight gear housings and to reinforce stressed areas in the high pressure rotor.

5.3 BENEFIT ASSESSMENT

Subsection 4.1.5 described the refined technology requirements for improvement of engine thermal and propulsive efficiency in the 2000 to 2010 time period. The paragraphs below present first, a description of the benefits of the thermal efficiency advancements, followed by a description of the benefits of the propulsive efficiency advancements. For consistency in comparing benefits, the reference engine and engines incorporating advanced technology features are scaled to the same cruise thrust for the two thrust classes shown in Table 4.1-I.

5.3.1 Advanced Channel Diffuser and Combustor

Table 5.3-I compares advanced diffuser/combustor aero-thermo technology to reference engine technology. As indicated in the table, diffuser and liner pressure losses are significantly lower while combustor exit temperature profiles are improved (hot spots and temperature gradients reduced). The advanced diffuser also reduces turbine cooling air temperature. This, along with the improved temperature profiles, allows a 2.6 percent reduction in turbine cooling air, as shown in Table 5.3-II.

TABLE 5.3-I
DIFFUSER/COMBUSTOR TECHNOLOGY COMPARISON

	Reference Engine	Advanced Diffuser/Combustor
Pressure Loss, percent		
Diffuser	1.9	1.0
Liner	2.5	2.0
Pattern Factor	0.37	0.25
First Turbine Blade Temperature Profile, °C (°F) (Max to Average)	107 (225)	65 (150)
Temperature Reduction at Constant Overall Pressure Ratio, °C (°F)		
Combustor Inner Diameter Feed	Base	-34 (-30)
Combustor Outer Diameter Feed	Base	-23 (-10)

TABLE 5.3-II
TURBINE COOLING AIR REDUCTION DUE TO ADVANCED DIFFUSER/COMBUSTOR

	Turbine Cooling Air Reduction (% Core Engine Flow)			
	First Stage Vanes	First Stage Blades	Second Stage Vanes	Second Stage Blades
Reference Engine Levels	7.40	2.75	1.30	0.30
-5.5 to -16.6 °C (-10 to -30°F) Cooling Air Temperature	0.25	0.20	0	0.05
-0.12 Pattern Factor	1.35	0	0.46	0
41.6 °C (-75°F) Blade Radial Temperature Profile	0	0.22	0	0.07
Totals	1.60	0.42	0.46	0.12

It should be noted that the second stage blade cooling flow of 0.30 is the minimum possible. While, theoretically, a 0.12 reduction is possible, its not practical. Thus, the first column in Table 5.3-III shows zero percent reduction for the second blade.

TABLE 5.3-III
THRUST SPECIFIC FUEL CONSUMPTION
IMPROVEMENT DUE TO ADVANCED DIFFUSER/COMBUSTOR

	Constant OPR (38.6)	Increased OPR (41) and Turbine Temperature
-1.4 Percent Pressure Loss	0.4	0.4
First Vane Cooling Air	0.15 (-1.6%)(1)	0.09 (-1.0%)(1)
First Blade Cooling Air	0.11 (-0.42%)	0.05 (-0.20%)
Second Vane Cooling Air	0.09 (-0.46%)	0.06 (-0.30%)
Second Blade Cooling Air	0 (0% at min flow)	0 (0%)
Improved Cycle	0	0.65
Total	0.75	1.25%

(1) Airfoil row cooling reduction as a percent of core airflow

The effects of these improvements on thrust specific fuel consumption are presented in Table 5.3-III. Two cases are included: the first assesses the benefits at constant overall pressure ratio; while the second uses some of the temperature profile and cooling air improvements to increase overall pressure ratio and turbine temperature, giving a larger thrust specific fuel consumption benefit. In the second case, the overall pressure ratio is increased to reflect the fact that the second blade is over-cooled at a 38.6 overall pressure ratio since, as previously noted, it is at minimum flow. By raising the overall pressure ratio to 41, the balance between cooling air temperature and the flow is re-established, providing the same blade life as the reference engine blade. The magnitude of the overall pressure ratio and turbine temperature increase is governed by material considerations. The second case was analyzed for the weight and cost comparison shown in Table 5.3-IV.

TABLE 5.3-IV
IMPACT OF ADVANCES ON
DIFFUSER/COMBUSTOR PERFORMANCE, WEIGHT AND COST
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced Combustor
TSFC, percent	Base	-1.25
Weight, kg (lb)	Base	36 (+80)
Cost, dollars	Base	+11,000
Maintenance Cost, $/EFH (1.25 hr flt)	Base	-1.60

The fuel burn advantage of the advanced technology diffuser/combustor is illustrated in Figure 5.3-1. This technology indicates a 1.6 percent improvement in quadjet fuel burn. Figures 5.3-2, -3, -4 present the DOC+I reduction potential of this concept for the three fuel prices. Since the primary benefit of the concept is improved fuel consumption, its relative benefit increases with fuel price.

5.3.2 Advanced Diffuser and Combustor Materials

The second portion of the diffuser/combustor technology benefit evaluation addressed advanced materials. Table 5.3-V compares the advanced technology diffuser/combustor materials with those used in the reference engine diffuser and combustor. Figure 5.3-5 shows how these advanced materials permit the large increase in overall pressure ratio. The thrust specific fuel consumption plot in this figure compares the contribution of advanced materials to that of advanced aero-thermo efficiency benefits. Included in the figure are the effects on high-pressure turbine efficiency caused by increasing the overall pressure ratio to 64:1. The high-pressure turbine efficiency shown in Table 4.1.VI was reduced by 0.7 percent to account for this.

TABLE 5.3-V
COMPARISON OF DIFFUSER/COMBUSTOR MATERIALS

	Reference Engine	Advanced
Liner Segment Support Frame	Forging	Lightweight Sheet
Liner Segments	B-1900 Castings	Ceramic Composite
Diffuser	Cast Inconel 718	Ceramic Composite
Outer Case	Inconel 718	Carbon Reinforced Advanced Nickel Alloy

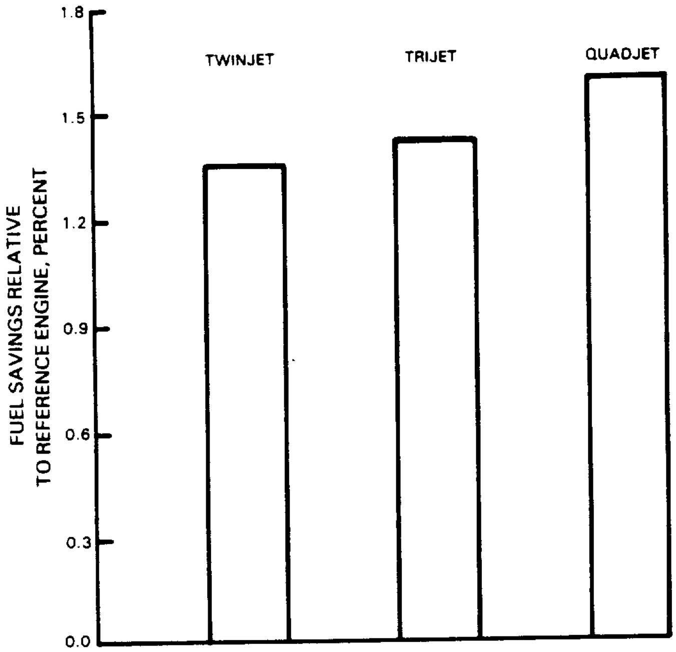

Figure 5.3-1 Fuel Savings Resulting from Improved Efficiency Design of the Advanced Diffuser/Combustor

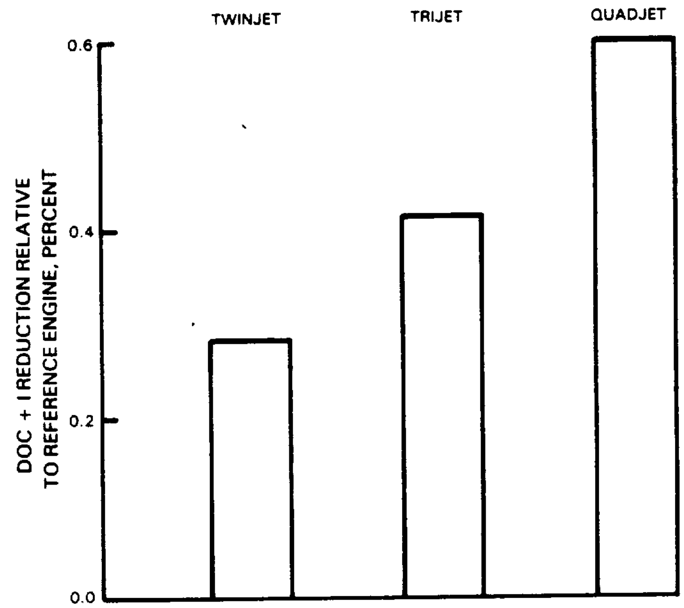

Figure 5.3-2 DOC+I Reduction Resulting from Improved Efficiency Design of the Advanced Diffuser/Combustor at \$0.26/liter (\$1.00/gallon) Fuel Cost

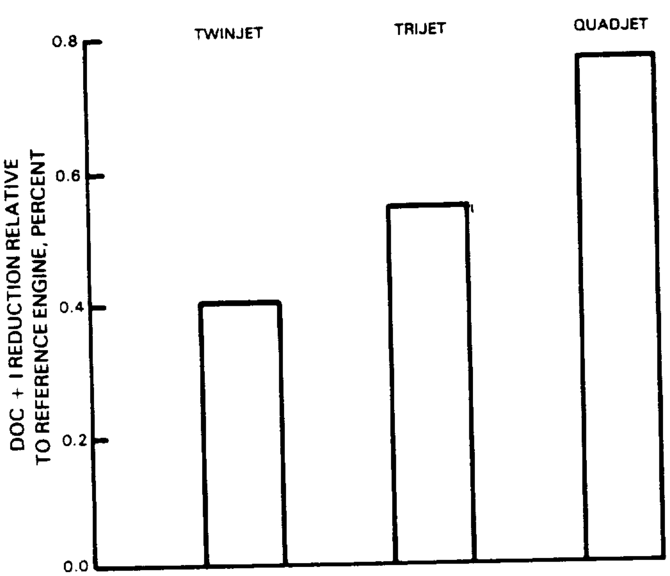

Figure 5.3-3 DOC+I Reduction Resulting from Improved Efficiency Design of the Advanced Diffuser/Combustor at \$0.40/liter (\$1.50/gallon Fuel Cost

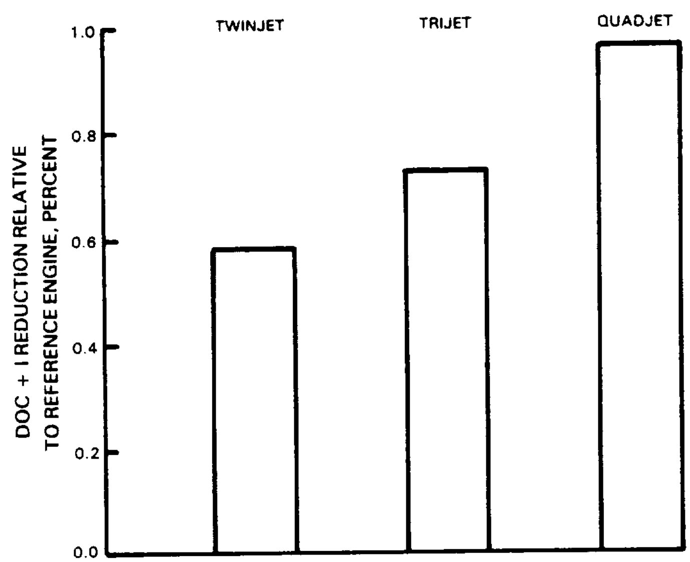

Figure 5.3-4 DOC+I Reduction Resulting from Improved Efficiency Design of the Advanced Diffuser/Combustor at \$0.66/liter (\$2.50/gallon) Fuel Cost

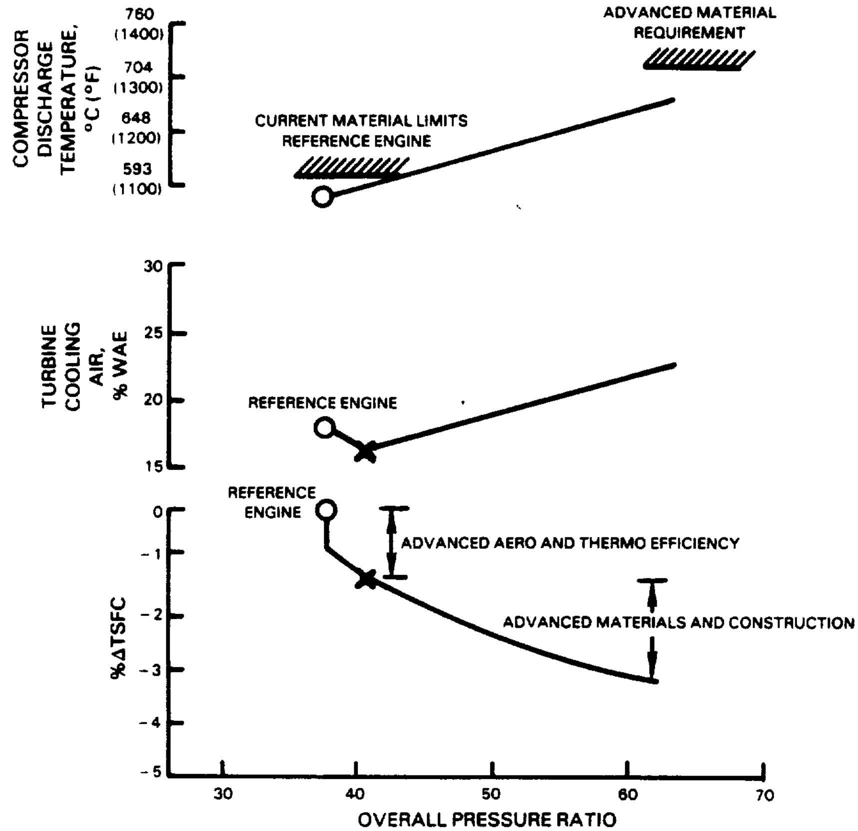

Figure 5.3-5 Benefits of Advanced Diffuser/Combustor Materials

Advanced combustor materials allow significant reductions in engine weight, engine first cost and maintenance cost, as shown in Table 5.3-VI. The weight and thrust specific fuel consumption improvements translate into a 2.7 percent fuel burn reduction for the quadjet airplane (Figure 5.3-6). This leads to the significant direct operating cost plus interest reductions shown in Figures 5.3-7, -8, -9.

TABLE 5.3-VI
IMPACT OF MATERIALS ADVANCES ON
DIFFUSER/COMBUSTOR PERFORMANCE, WEIGHT AND COST
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced
TSFC, percent	Base	-1.9
Weight, kg (lb)	Base	-54 (-120)
Cost, dollars	Base	-22,000
Maintenance Cost, $/EFH (1.25 hr flt)	Base	-3.70

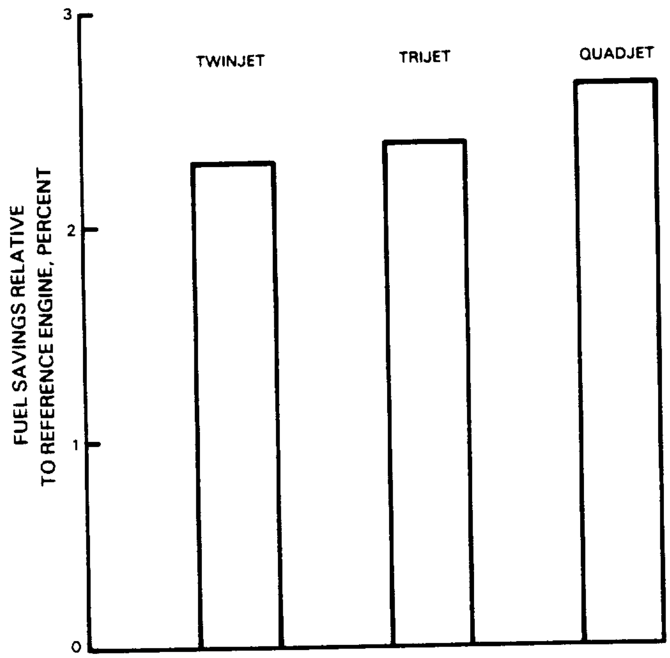

Figure 5.3-6 Fuel Savings Resulting from Advanced Materials in the Diffuser/Combustor

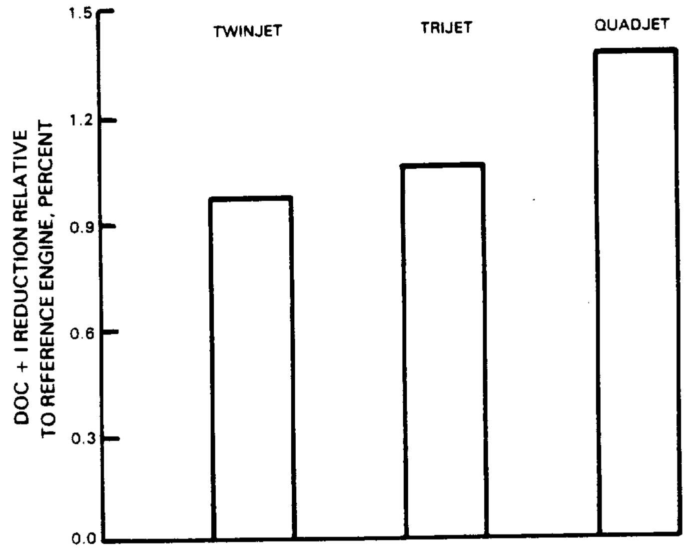

Figure 5.3-7 DOC+I Reduction Resulting from Advanced Materials in the Diffuser/Combustor at $0.26/liter ($1.00/gallon) Fuel Cost

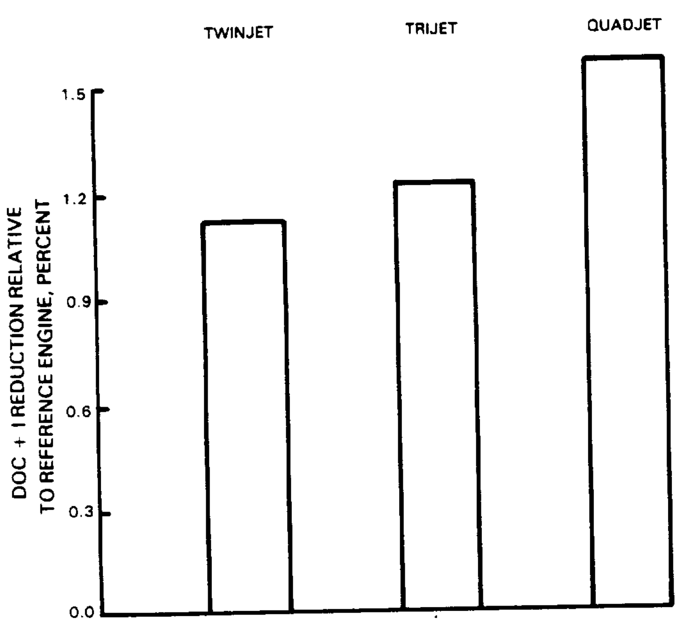

Figure 5.3-8 DOC+I Reduction Resulting from Advanced Materials in the Advanced Diffuser/Combustor at $0.40/liter ($1.50/gallon) Fuel Cost

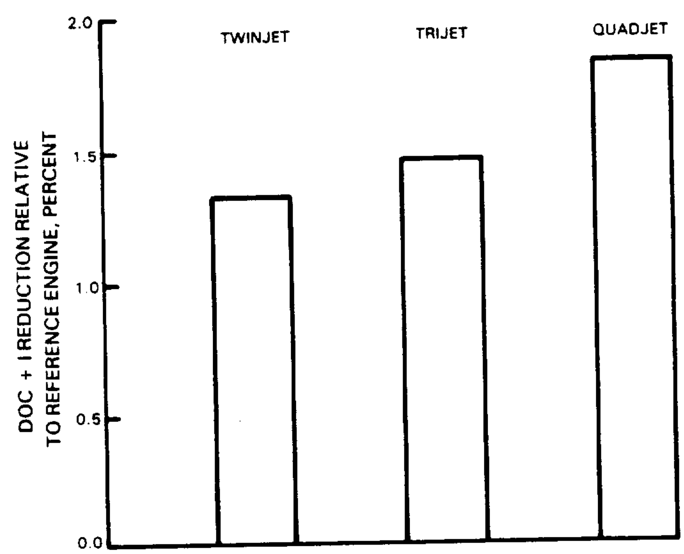

Figure 5.3-9 DOC+I Reduction Resulting from Advanced Materials in the Advanced Diffuser/Combustor at $0.66/liter ($2.50/gallon) Fuel Cost

5.3.3 High Efficiency High Pressure Turbine

Two advanced technology high pressure turbines were evaluated, one with metallic vanes, the other with ceramic vanes. Each presumed the incorporation of the advanced diffuser/combustor and its improved materials. Both configurations offer large thrust specific fuel consumption improvements over the high pressure turbine of the reference engine due to their improved aerodynamics, better sealing, smaller clearances, and more efficient cooling. The higher temperature capability of the ceramic vanes permits lower cooling flows, giving them a small (0.4 percent) thrust specific fuel consumption advantage over the advanced metallic vane. As shown in Table 5.3-VII, the ceramic vanes also have small weight and cost advantages over the metallic vanes.

TABLE 5.3-VII
COMPARISON OF HIGH PRESSURE TURBINES
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced High Pressure Turbines	
		Metallic Vanes	Ceramic Vanes
TSFC, percent	Base	-3.05	-3.45
Weight, kg (lb)	Base	-24 (-55)	-27 (-60)
Cost, dollars	Base	-19K	-27,000
Maintenance Cost, $/EFH (1.25 hr flt)	Base	-2.00	-2.00

These performance advantages translate into the significant fuel burn reductions indicated in Figure 5.3-10. Benefits of 4.1 percent with metallic vane technology and 4.6 percent with ceramic vanes are achieved in fuel burn for the quadjet airplane. DOC+I benefits for both concepts are also significant (Figures 5.3-11, -12, and -13). Advanced metallic vane technology offers the potential of a 2.3 percent DOC+I reduction on the quadjet airplane at $0.40/liter ($1.50 per gallon) fuel, while ceramic vane technology offers about 2.6 percent.

5.3.4 High Efficiency Compressors

Two distinct compressor configurations were evaluated: an advanced all-axial compressor suitable for a large high pressure ratio engine; and an axial-centrifugal compressor prompted by concerns that, at the high pressure ratios required for optimum performance (55 overall pressure ratio for the small engine), the back stage blades of an all-axial compressor in the small engine would be too small to provide the desired levels of efficiency when erosion and tip-clearance-to-blade-span ratio are considered.

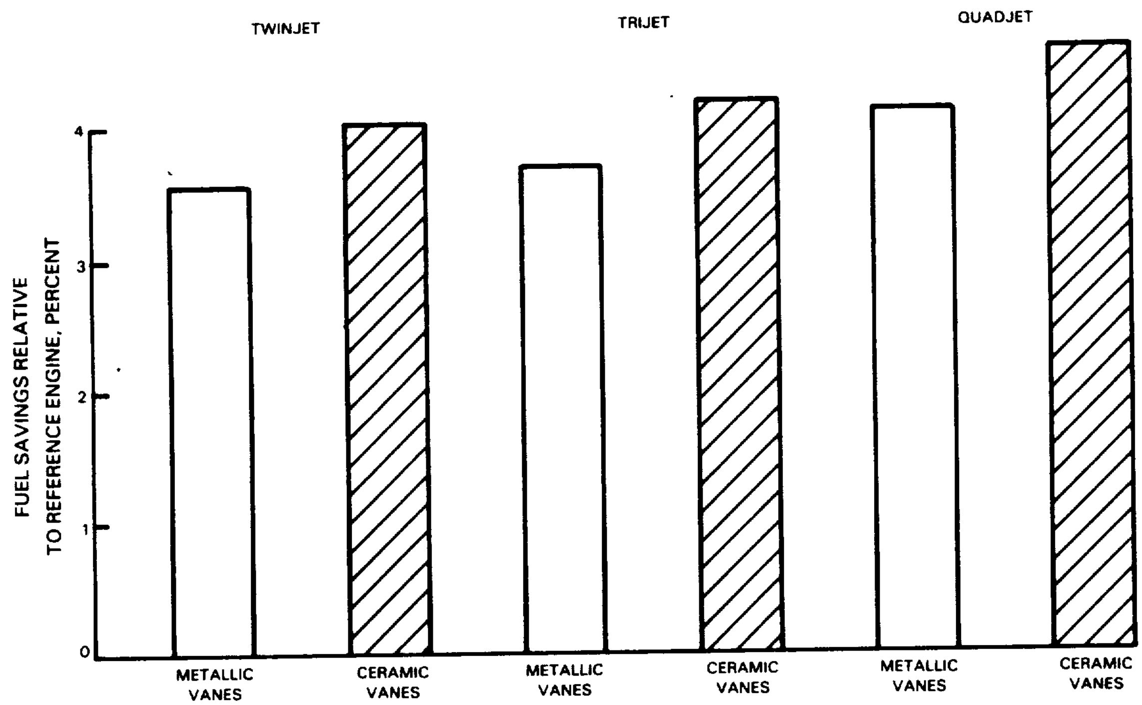

Figure 5.3-10 Comparison of Fuel Savings for the Advanced Metallic Vane and Ceramic Vane High Pressure Turbines

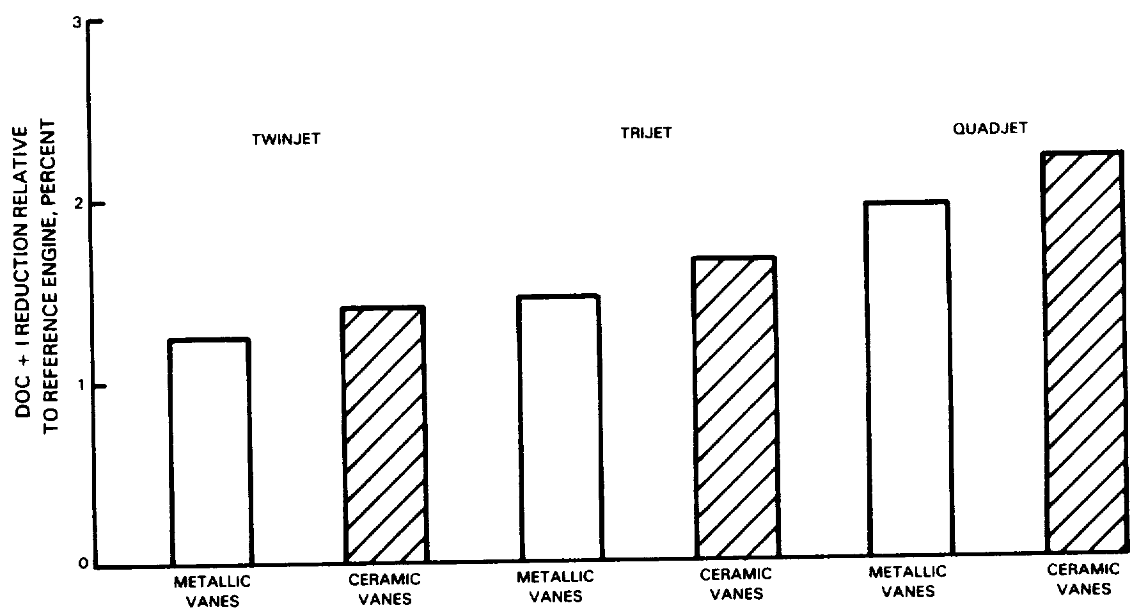

Figure 5.3-11 Comparison of DOC+I Reduction for the Advanced Metallic Vane and Ceramic Vane High Pressure Turbines at $0.26/liter ($1.00/gallon) Fuel Cost

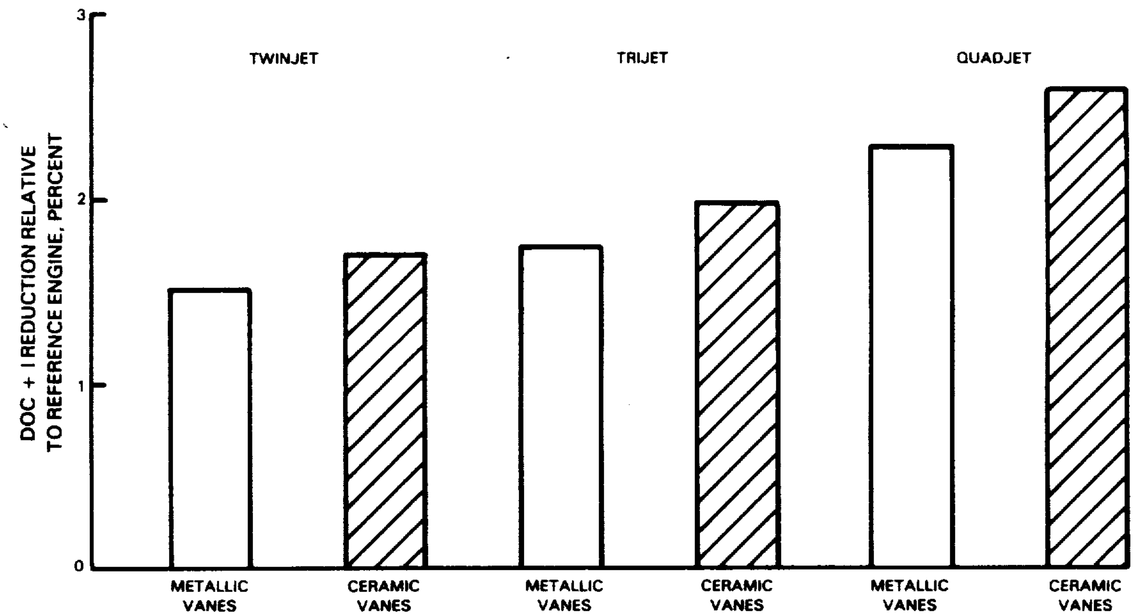

Figure 5.3-12 Comparison of DOC+I Reduction for the Advanced Metallic Vane and Ceramic Vane High Pressure Turbines at $0.40/liter ($1.50/gallon) Fuel Cost

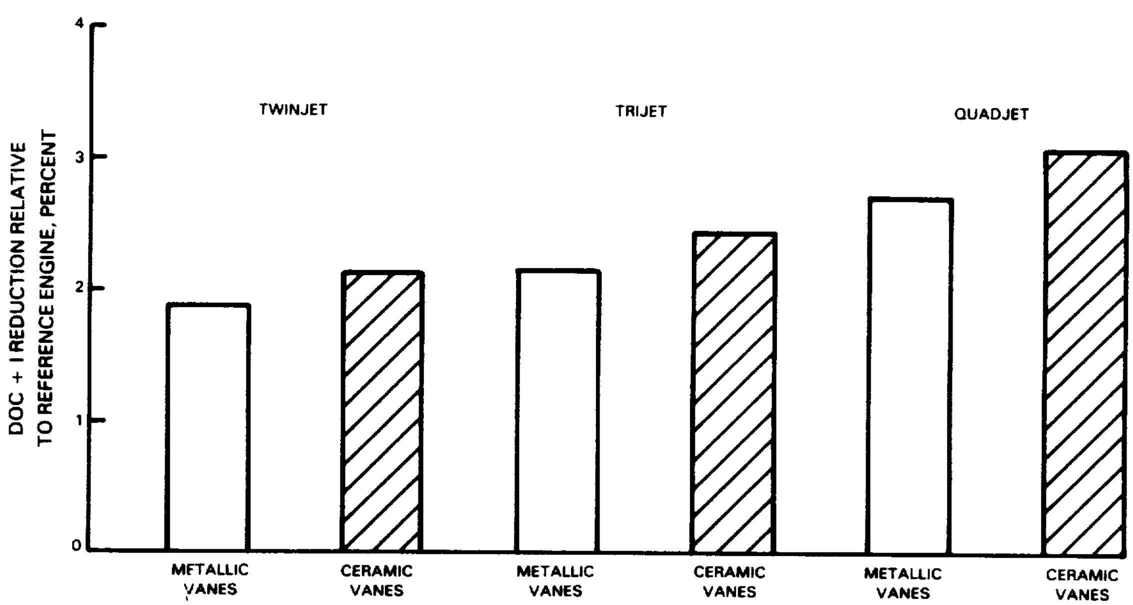

Figure 5.3-13 Comparison of DOC+I Reduction for the Advanced Metallic Vane and Ceramic Vane High Pressure Turbines at $0.66/liter ($2.50/gallon) Fuel Cost

Comparison of an advanced technology all-axial compressor to reference engine technology is shown in Table 5.3-VIII. The comparison was made at equal thrust size and equal pressure ratio, so the thrust specific fuel consumption advantage reflects aerodynamic improvements, not cycle changes. Maintenance cost advantage of the advanced compressor results primarily from improving the tolerance to deterioration and reducing the number of blades.

TABLE 5.3-VIII
ALL AXIAL COMPRESSOR COMPARISON
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced Axial
Overall Polytropic Efficiency, percent	Base	+1.6
TSFC, percent	Base	-1.4
Weight, kg (lb)	Base	-22 (-50)
Cost, dollars	Base	+400
Maintenance Cost, $/EFH (1.25 hr flt)	Base	-4.00

An advanced technology axial-centrifugal compressor is also compared to the all axial reference engine compressor in Table 5.3-IX. As expected, the axial-centrifugal configuration shows less thrust specific fuel consumption advantage over the reference engine than the advanced all-axial compressor. It does, however, have larger airplane cost and maintenance cost advantages. Again, the comparisons are made at equal thrust and pressure ratio. A smaller thrust size (21,351 N (4800 lb) at cruise vs. 51,332 N (11,540 lb) for all-axial comparison) was used in this comparison because the small engine is more likely to require an axial-centrifugal compressor.

TABLE 5.3-IX
AXIAL-CENTRIFUGAL COMPRESSOR COMPARISON
(10,668 m (35,000 ft); 0.8 Mn; 21,351 N (4800 lb) Max Cruise Thrust)

	Reference Engine	Advanced Axial-Centrifugal
TSFC, percent	Base	-1.0
Weight, kg (lb)	Base	27 (+60)
Cost, dollars	Base	-40,000
Maintenance Cost, $/EFH (1.25 hr flt)	Base	-6.00

Figure 5.3-14 shows the potential fuel burn benefits for both types of advanced technology compressors, each relative to the reference engine. The axial-centrifugal compressor has less fuel burn advantage, since it had less thrust specific fuel consumption advantage and a slightly higher weight than the all-axial.

However, that advantage is diminished in a DOC+I comparison of the two compressor types with fuel prices at $0.26, $0.40 and $0.66/liter, ($1.00, $1.50 and $2.50/gallon), (Figures 5.3-15, -16 and -17, respectively). Up to a fuel price of $0.40/liter ($1.50 a gallon), the axial-centrifugal compressor has a higher DOC+I reduction advantage over the reference engine than the all-axial compressor has. This is caused by the airplane cost and maintenance cost advantages of the axial-centrifugal which overshadow its lesser performance advantage in the relatively fuel insensitive short range twinjet. Thrust requirements of the twinjet (97 - 111,205 N (22 - 25,000 lb) takeoff) fall into the small engine class, while the trijet and quadjet require 177,928 N (40,000+ lb) of takeoff thrust, putting them in the large engine category. Thus, the axial-centrifugal results are most germane to the twinjet, while the all-axial results are most germane to the trijet and quadjet.

5.3.5 Active Clearance Control

The advanced technology active clearance control is a closed loop system that controls blade-to-case clearances in the compressor and high and low pressure turbines. By keeping clearances tighter than would be possible with reference engine technology, the advanced system reduces thrust specific fuel consumption by 1 percent, although causing a slight weight increase (Table 5.3-X). Figure 5.3-18 shows that this converts into a 1.3 percent decrease in fuel burn on the quadjet. Figures 5.3-19, -20, and -21 present the DOC+I reduction for the three different fuel prices.

TABLE 5.3-X
COMPARISON OF ACTIVE CLEARANCE CONTROL
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced Active Clearance Control
TSFC, percent	Base	-1.0
Weight, kg (lb)	Base	+13.6 (+30)
Cost, dollars	Base	0
Maintenance Cost, $/EFH (1.25 hr flt)	Base	0

5.3.6 High Efficiency Low Pressure Turbine

Improvements in low pressure turbine technology offer significantly less thrust specific fuel consumption reduction than offered by improvements to the high pressure turbine. The potential for weight and cost reductions (shown in Table 5.3-XI), however, is roughly comparable to that of the high pressure turbine. Since thrust specific fuel consumption is the dominant factor in fuel burn, Figure 5.3-22 shows advanced low pressure turbine technology to have substantially less fuel burn benefit potential (0.7 percent in the quadjet) than did the advanced high pressure turbine technology.

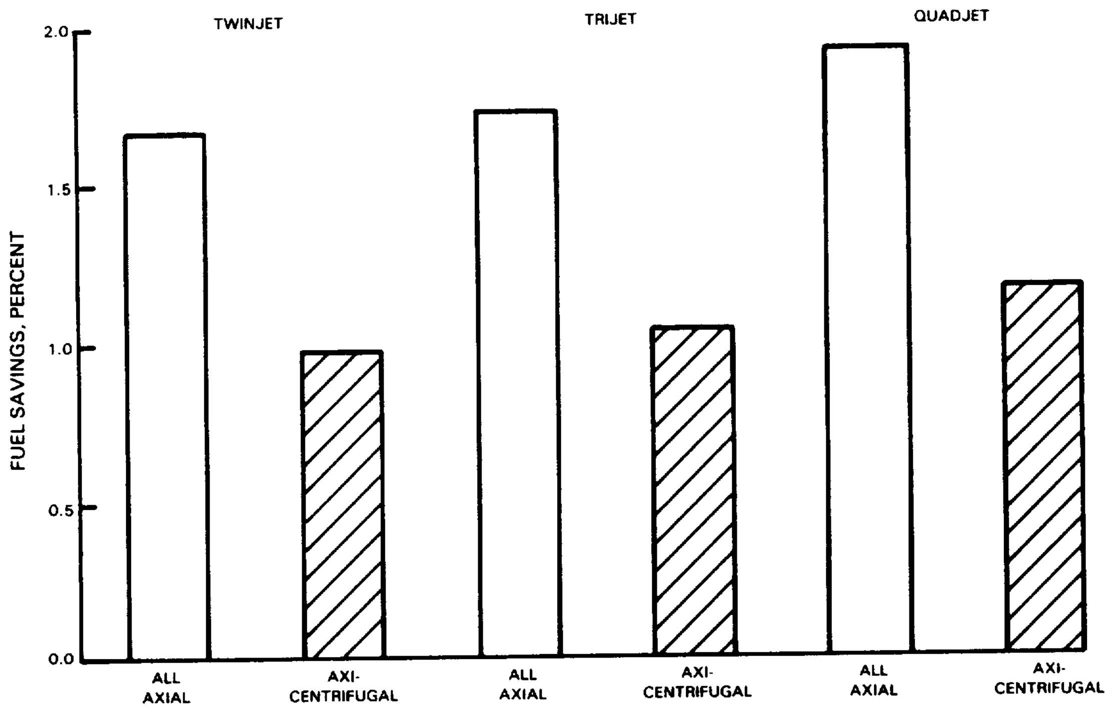

Figure 5.3-14 Comparison of Fuel Savings for the All Axial and the Axial-Centrifugal Compressors

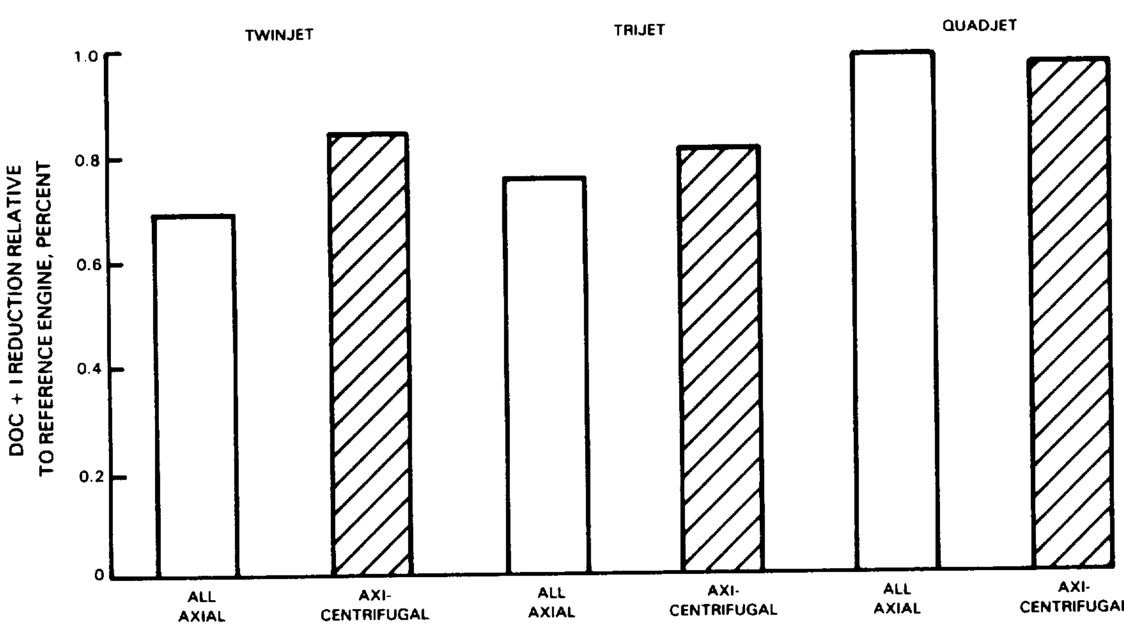

Figure 5.3-15 Comparison of DOC+I Reduction for the All Axial and the Axial-Centrifugal Compressors at $0.26/liter ($1.00/gallon) Fuel Cost

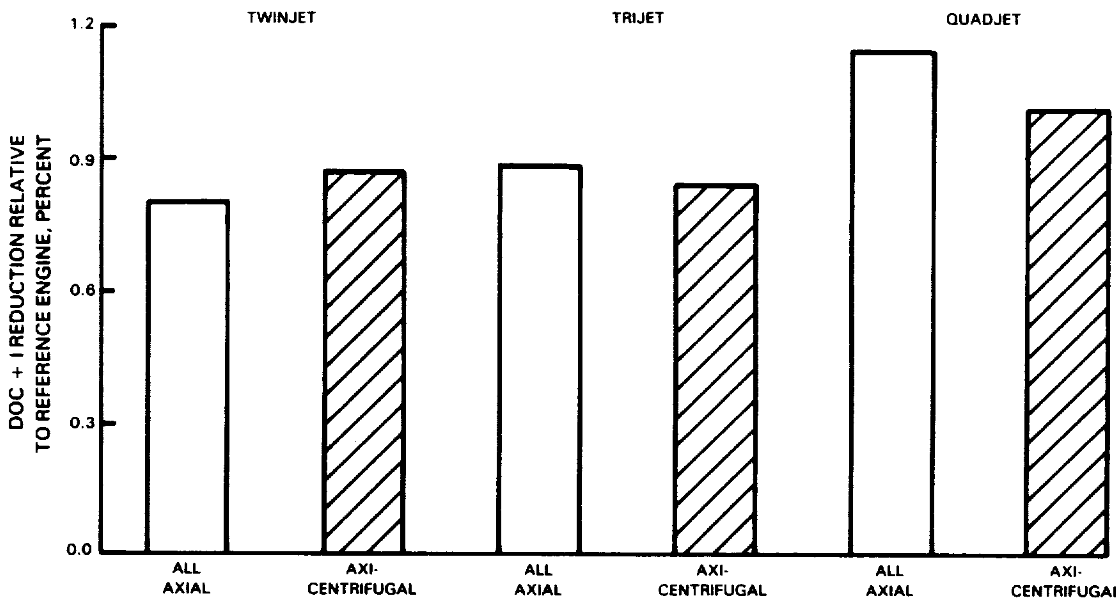

Figure 5.3-16 Comparison of DOC+I Reduction for the All Axial and the Axial-Centrifugal Compressors at $0.40/liter ($1.50/gallon) Fuel Cost

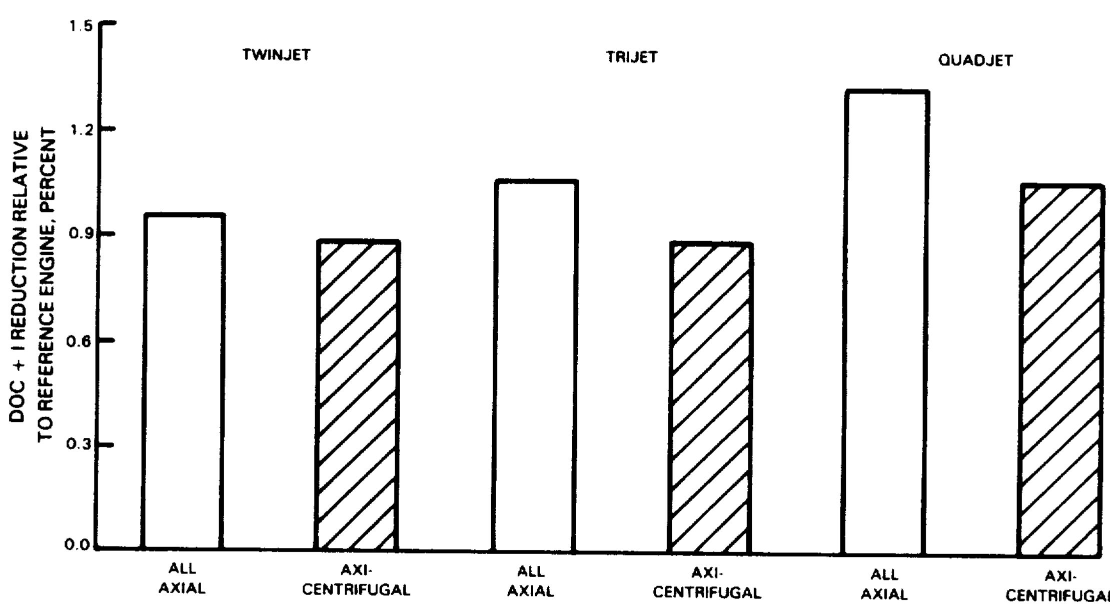

Figure 5.3-17 Comparison of DOC+I Reduction for the All Axial and the Axial-Centrifugal Compressors at $0.66/liter ($2.50/gallon) Fuel Cost

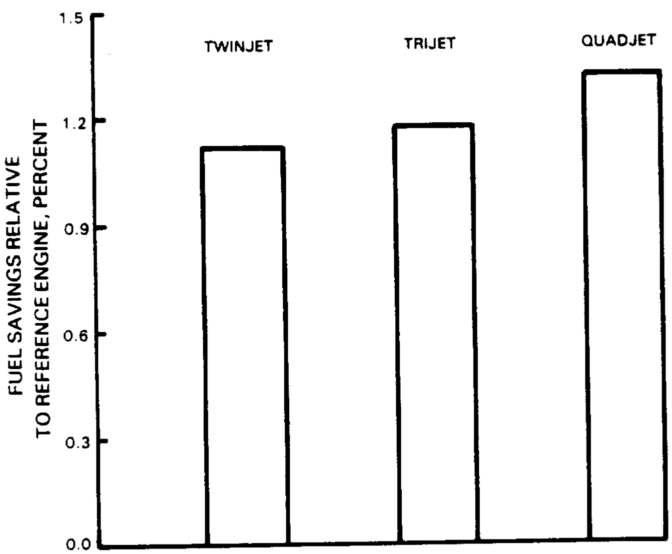

Figure 5.3-18 Fuel Savings from Advanced Technology Active Clearance Control

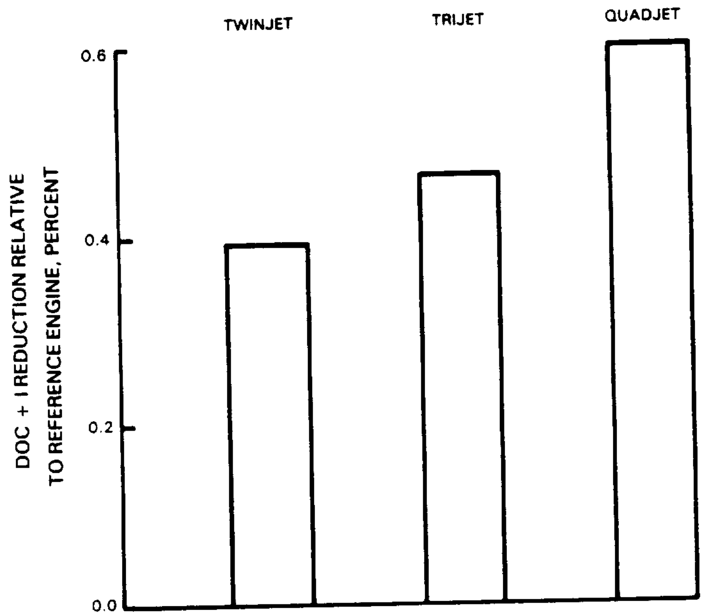

Figure 5.3-19 DOC+I Reduction Resulting from Advanced Technology Active Clearance Control at $0.26/liter ($1.00/gallon) Fuel Cost

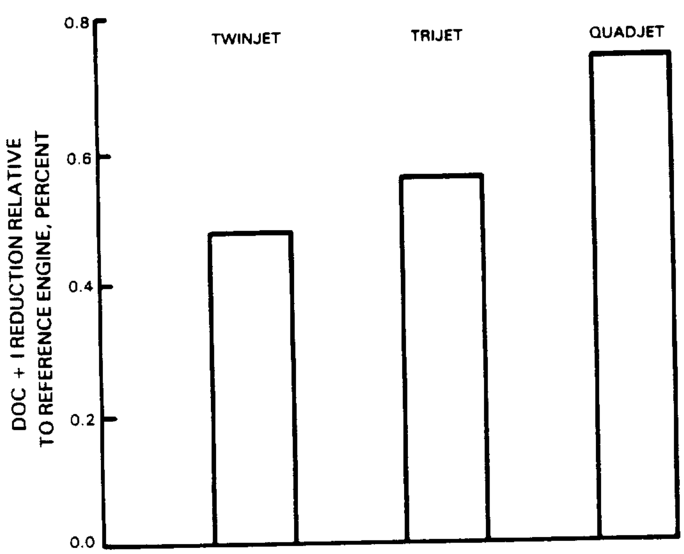

Figure 5.3-20 DOC+I Reduction Resulting from Advanced Technology Active Clearance Control at $0.40/liter ($1.50/gallon) Fuel Cost

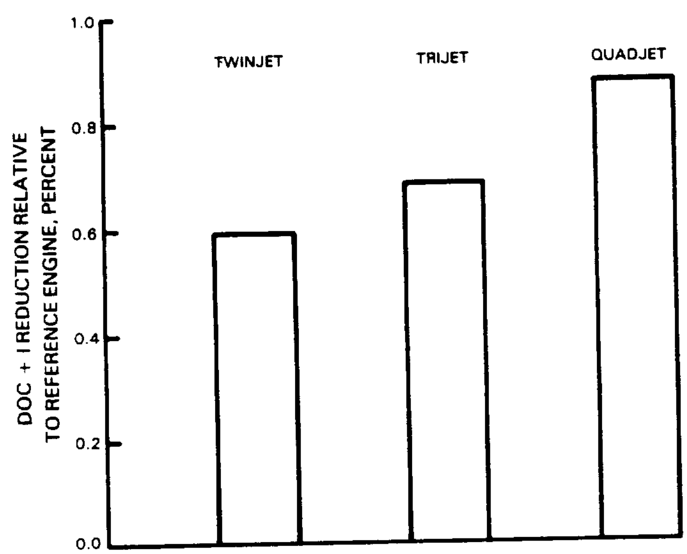

Figure 5.3-21 DOC+I Reduction Resulting from Advanced Technology Active Clearance Control at $0.66/liter ($2.50/gallon) Fuel Cost

TABLE 5.3-XI
COMPARISON OF LOW PRESSURE TURBINES
(10,668 m (35,000 ft); 0.8 Mn; 51,332 N (11,540 lb) Max Cruise Thrust)

	Reference Engine	Advanced Low Pressure Turbine
TSFC, percent	Base	-0.5
Weight, kg (lb)	Base	-22 (-50[1])
Cost, dollars	Base	-30,000[1]
Maintenance Cost, \$/EFH (1.25 hr flt)	Base	-1.00[1]

(1) Considers Differences in Rotor Construction Only

Direct operating cost plus interest benefits are shown in Figure 5.3-23, -24, -25. A potential reduction of 0.5 percent is seen in the quadjet airplane at \$0.40/liter (\$1.50 per gallon) fuel price.

5.3.7 Advanced Swept Fan

Three types of geared drive fans were investigated: shrouded and unshrouded geared versions of the reference engine fan and an advanced three dimensional swept fan. As shown in Figure 5.3-26, all were superior in efficiency to the direct drive reference engine fan. Direct drive requires the fan to run at a higher (and less optimum) speed than the geared fans to optimize integrated performance.

Also indicated in Figure 5.3-26 is the advantage to fan efficiency of reducing fan pressure ratio. Figure 5.3-27 shows how this fan efficiency trend translates into thrust specific fuel consumption reduction and the advantages of the shroudless, geared fan and the swept fan over the shrouded, geared fan. These trends assume the use of the advanced technology nacelles and other advanced technologies. While a low fan pressure ratio/high bypass ratio provides a thrust specific fuel consumption advantage, the difference in performance for the study fans is greatest at the high fan pressure ratio.

Table 5.3-XII compares the performance, weight and cost benefits of the shroudless, geared fan and the advanced swept fan to the geared, shrouded fan at high and low pressure ratios. This table includes the effects of scaling the engines to constant thrust sizes, which benefits the better performing shroudless and swept fans. Maintenance cost of the shroudless and swept fans is higher mainly because they are less repairable than the shrouded fan because they are hollow and have a limited leading edge thickness available for blending damaged areas.

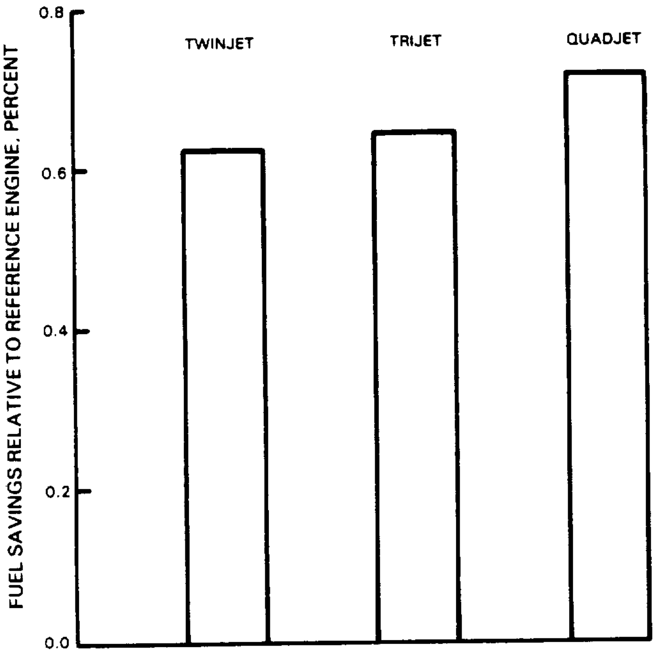

Figure 5.3-22 Fuel Savings from Advanced Technology Low Pressure Turbine

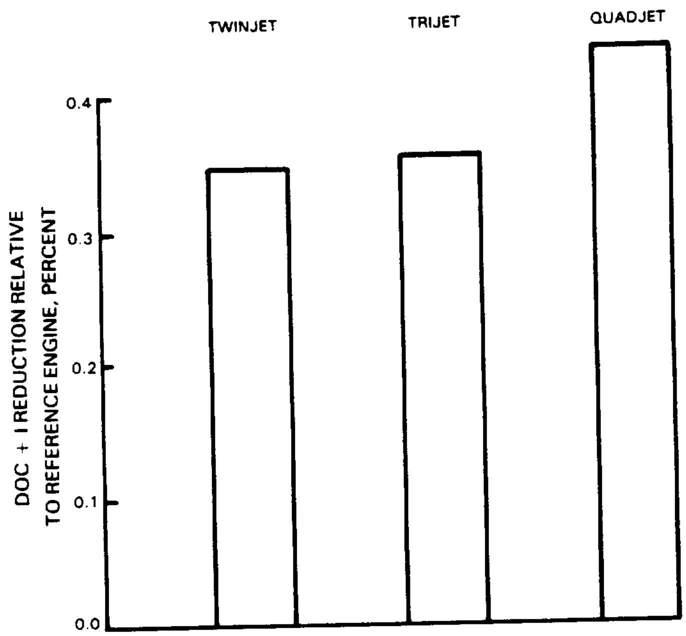

Figure 5.3-23 DOC+I Reduction Resulting from Advanced Technology Low Pressure Turbine at $0.26/liter ($1.00/gallon) Fuel Cost

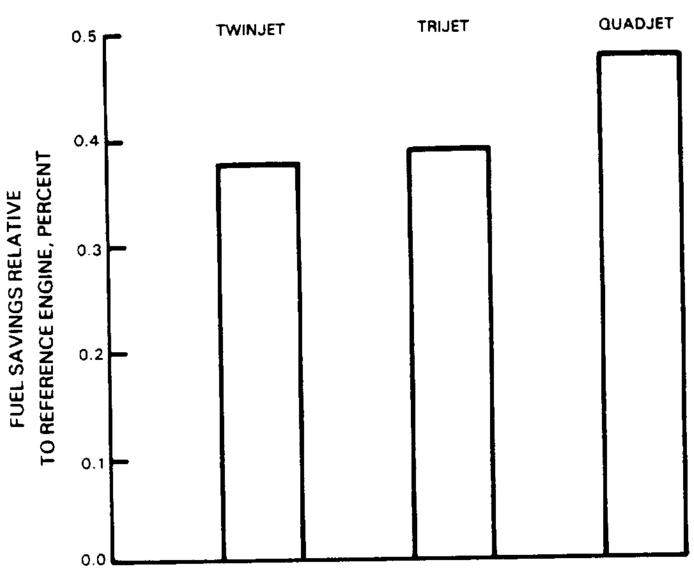

Figure 5.3-24 DOC+I Reduction Resulting from Advanced Technology Low Pressure Turbine at $0.40/liter ($1.50/gallon) Fuel Cost

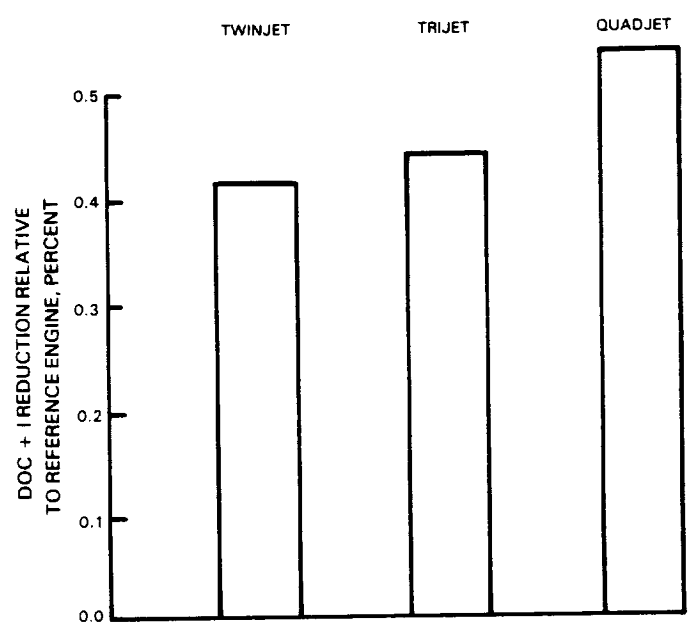

Figure 5.3-25 DOC+I Reduction Resulting from Advanced Technology Low Pressure Turbine at $0.66/liter ($2.50/gallon) Fuel Cost

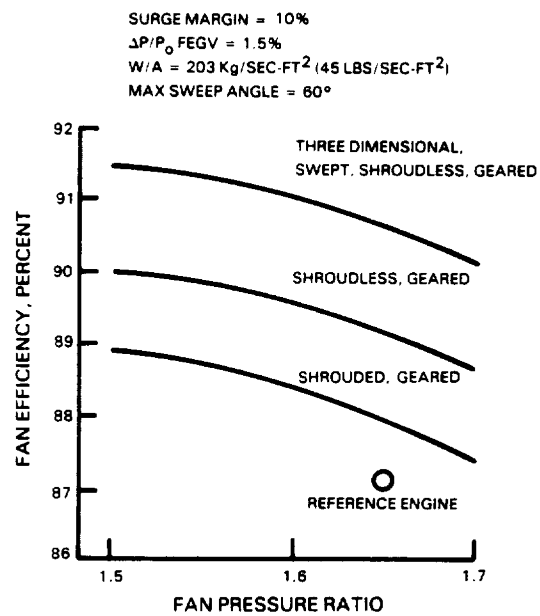

Figure 5.3-26 Efficiency Comparison of Three Study Fans to Reference Engine

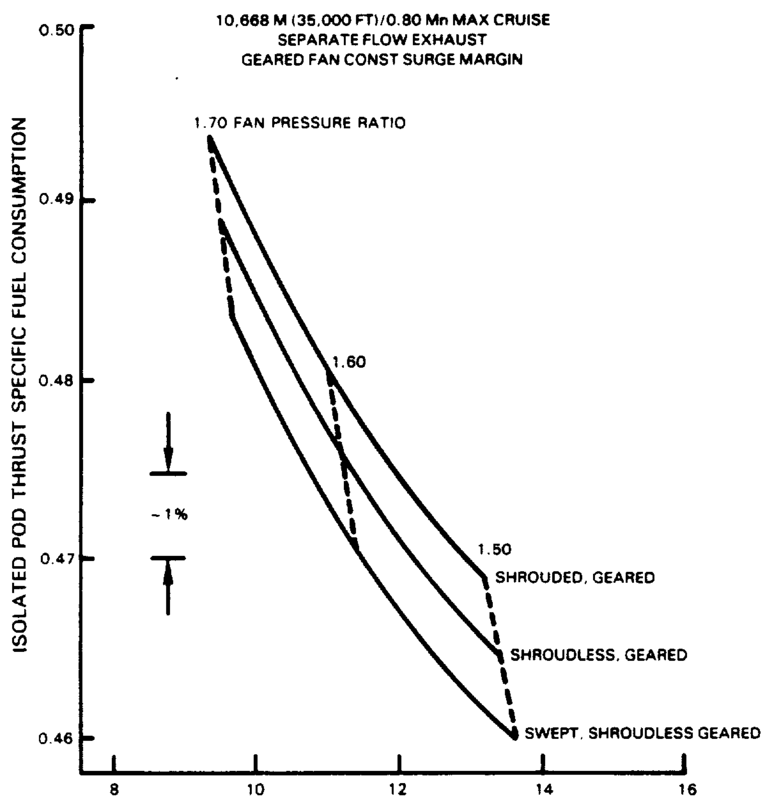

Figure 5.3-27 Effect of Fan Pressure Ratio and Bypass Ratio on Thrust Specific Fuel Consumption

TABLE 5.3-XII

PERFORMANCE, WEIGHT AND COST COMPARISONS OF THE THREE STUDY FANS
(10,668 m (35,000 ft); 0.8Mn; 48,930 N (11,000 lb) Max Cruise Thrust)

	1.5 FPR			1.7 FPR		
	Shrouded	Shroudless	Swept	Shrouded	Shroudless	Swept
TSFC, percent	Base	-0.85	-1.9	Base	-0.95	-2.05
Weight, kg (lb)	Base	0	-18 (-40)	Base	0	-13 (-30)
Cost, dollars	Base	0	+10,000	Base	+20,000	+30,000
Maintenance Cost, $/EFH (1.25hr flt)	Base	+3.90	+4.25	Base	+3.00	+3.25

Using influence coefficients, thrust specific fuel consumption and weight can be translated into the mission fuel burn comparisons presented in Figure 5.3-28. As would be expected, the long range quadjet airplane shows the most benefit from the advanced swept fan, about 2.6 percent at 1.5 fan pressure ratio; followed by the medium range trijet at 2.3 percent; and the twinjet at 2.2 percent. Including engine cost and maintenance cost gives the DOC+I trends shown in Figure 5.3-29. Again, the quadjet shows the greatest benefit from the advanced swept fan, a 1.2 percent improvement in DOC+I at 1.5 fan pressure ratio and $0.40/liter ($1.50 per gallon) fuel; with the trijet at 0.9 percent and the twin at 0.7 percent.

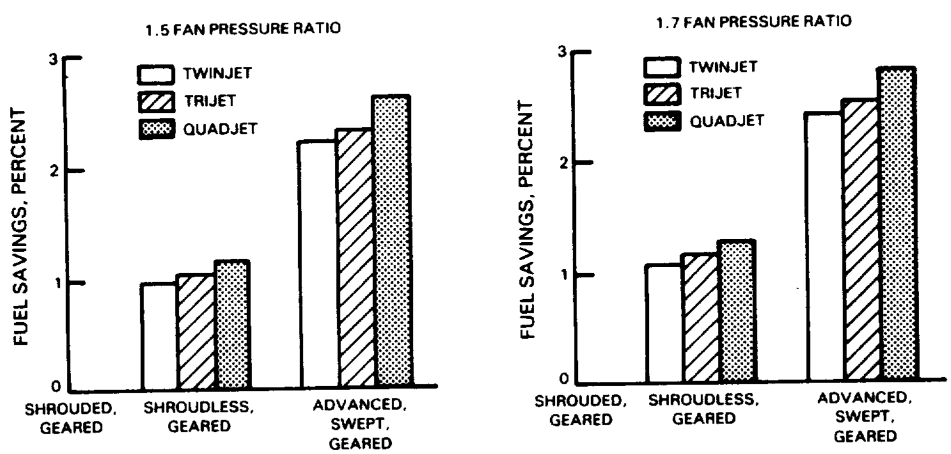

Figure 5.3-28 Comparison of Fuel Savings for Shroudless and Swept Geared Fans over Shrouded, Geared Fan at Different Pressure Ratios

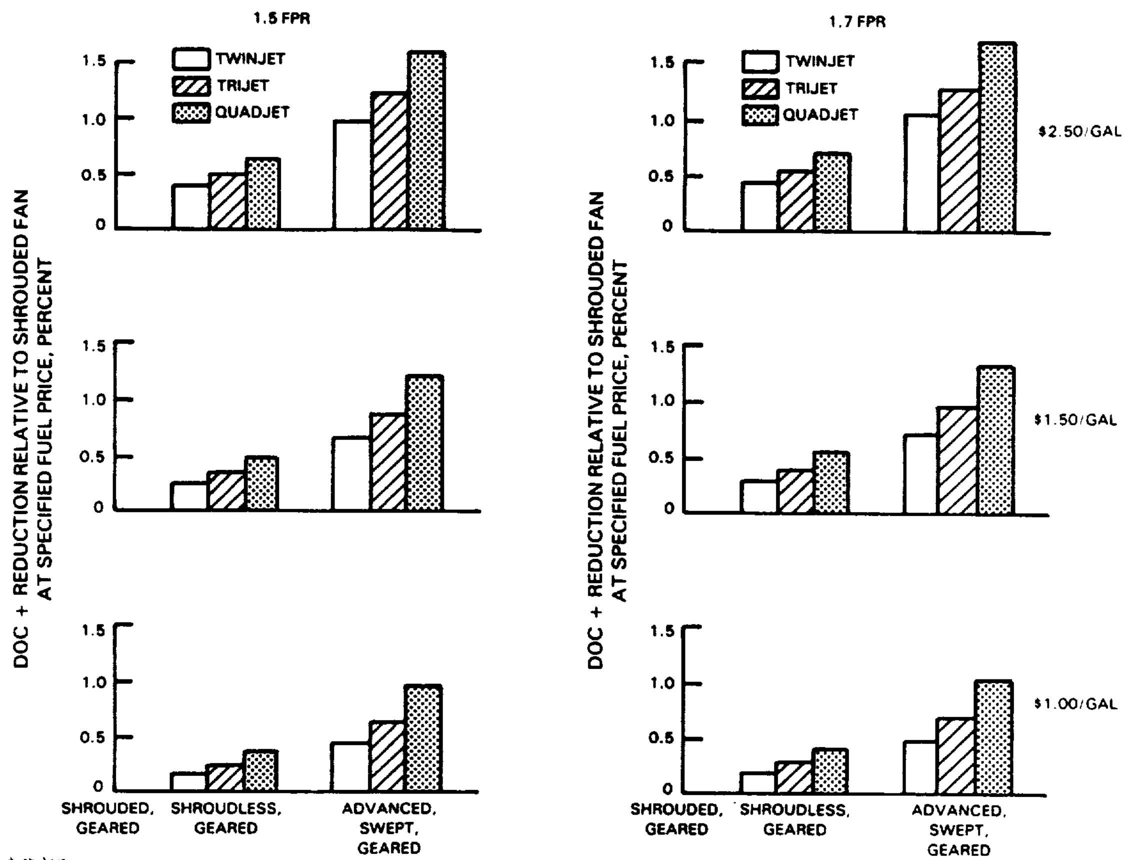

Figure 5.3-29 Comparison of DOC+I Reduction for Shroudless and Swept Geared Fans over Shrouded, Geared Fan at Different Pressure Ratios

5.3.8 Advanced Geared Low Pressure Spool

The advanced technology geared low pressure spool offers the potential for significant fuel burn and economic benefits relative to direct drive configurations. These benefits were difficult to isolate since they were tied to cycle selection and to technology levels of the other components. To overcome this problem, the cycle studies were expanded to include weight and cost. In this manner, a full benefit analysis could be performed for the advanced geared low pressure spool. Table 5.3-XIII presents a matrix of the cycles included in the analysis, comparing the reference engine to the direct drive, mixed and separate flow configurations. Table 5.3-XIV compares the reference engine to the geared drive configurations. All configurations, except the reference engine, use the advanced technology concepts described previously. Installed thrust specific fuel consumption and propulsion weights are shown for both the reference engine and advanced technology nacelles.

Propulsion system performance was calculated for all points in the matrix. Weights and costs were calculated in some detail for several configurations and crossplotted to estimate weights and costs for the rest of the matrix. Influence coefficients were then used to calculate fuel burn and direct operating cost plus interest benefits relative to the reference engine for all configurations.

TABLE 5.3-XIII
COMPARISON OF DIRECT DRIVE LOW PRESSURE SPOOL OPERATING PARAMETERS

	Reference Engine	Direct Drive Separate Flow	Direct Drive Separate Flow	Direct Drive Separate Flow	Direct Drive Mixed Flow	Direct Drive Mixed Flow
Bypass Ratio	7.2	7.0	9.6	12.8	7.0	9.0
Fan Pressure Ratio	1.65	1.88	1.70	1.50	1.86	1.68
Stages	1-4-10 -2-5	1-4-11 -2-5	1-5-11 -2-7	1-5-11 -2-7	1-4-11 -2-5	1-5-11 -2-7
Takeoff Thrust, N (lb)	178,128 (40,045)	204,617 (46,000)	223,522 (50,250)	245,540 (55,200)	202,837 (45,600)	222,187 (49,950)
Max Cruise Thrust, N (lb)	41,457 (9320)	48,930 (11,000)	50,709 (11,400)	51,154 (11,500)	48,485 (10,900)	50,220 (11,290)
Design W_{AT2}, kg/sec (lb/sec)	679 (1498)	689 (1520)	913 (2015)	1,184 (2612)	686 (1513)	857 (1890)
Installed Cruise TSFC (Current Nacelle)		0.535	0.505	0.498	0.512	0.497
Installed Cruise TSFC (Advanced Nacelle)	0.548	0.526	0.493	0.484	0.511	0.494
Engine Weight, kg (lb)	3,549 (7826)	2,702 (5959)	3,356 (7400)	4,141 (9130)	2,771 (6110)	3,288 (7250)
Nacelle & Pylon Weight, kg (lb) (Current Nacelle)		1,682 (3710)	2,131 (4700)	2,592 (5715)	2,347 (5175)	2,739 (6040)
Nacelle & Pylon Weight, kg (lb) (Advanced Nacelle)	1,603 (3535)	1,378 (3040)	1,746 (3850)	2,122 (4680)	1,923 (4240)	2,245 (4950)
Engine Cost, 1981 ($1000)	Base	-447	-151	+264	-417	-175
Eng. Maintenance Cost, 1981$/EFH (1.25 hr/flt)	Base	-32.9	-19.4	-14.8	-32.9	-20.4

TABLE 5.3-XIV
COMPARISON OF GEARED DRIVE LOW PRESSURE SPOOL OPERATING PARAMETERS

	Reference Engine	Geared Drive				
		Separate Flow	Separate Flow	Separate Flow	Mixed Flow	Mixed Flow
Bypass Ratio	7.2	9.0	12.8	17.0	9.0	12.8
Fan Pressure Ratio	1.65	1.75	1.53	1.40	1.70	1.50
Stages	1-4-10 -2-5	1-3-11 -2-5	1-3-11 -2-5	1-3-11 -2-5	1-3-11 -2-5	1-3-11 -2-5
Takeoff Thrust, N (lb)	178,128 (40,045)	221,520 (49,800)	257,105 (57,800)	278,902 (62,700)	225,968 (50,800)	259,107 (58,250)
Max Cruise Thrust, N (lb)	41,457 (9320)	50,264 (11,300)	53,600 (12,050)	55,380 (12,450)	51,065 (11,480)	53,155 (11,950)
Design W_{AT2}, kg/sec (lb/sec)	679 (1498)	857 (1890)	1,184 (2612)	1,544 (3404)	857 (1890)	1,184 (2612)
Installed Cruise TSFC (Current Nacelle)		0.503	0.477	0.467	0.488	0.474
Installed Cruise TSFC (Advanced Nacelle)	0.548	0.491	0.463	0.451	0.485	0.468
Engine Weight, kg (lb)	3,549 (7826)	2,936 (6475)	3,503 (7725)	4,329 (9545)	2,936 (6475)	3,503 (7725)
Nacelle & Pylon Weight, kg (lb) (Current Nacelle)		1,800 (3970)	2,313 (5100)	2,771 (6110)	2,458 (5420)	3,220 (7100)
Nacelle & Pylon Weight, kg (lb) (Advanced Nacelle)	1,603 (3535)	1,474 (3250)	1,895 (4178)	2,270 (5005)	2,013 (4440)	2,637 (5815)
Engine Cost, 1981 ($1000)	Base	-387	-177	+183	-387	-177
Eng. Maintenance Cost, 1981$/EFH (1.25 hr/flt)	Base	-27.9	-23.4	-19.3	-27.9	-23.4

Fuel burn and DOC+I benefits for advanced technology engines are shown in Figure 5.3-30 as functions of bypass ratio. These curves assume use of advanced technology nacelles on the long range quadjet airplane. Advanced technology direct drive engines reach maximum fuel burn reductions at a bypass ratio of about 11 and maximum DOC+I benefit at a bypass ratio of about 10. Maximum benefits for advanced geared configurations occur at bypass ratios of about 17 to 18 for fuel burn and 13 for DOC+I. Comparing the geared configuration with a bypass ratio of 13 to the direct drive engine with a bypass ratio of 10 (both separate flow) shows a fuel burn advantage of about 7 percent and a DOC+I advantage of about 4 percent for the geared engine.

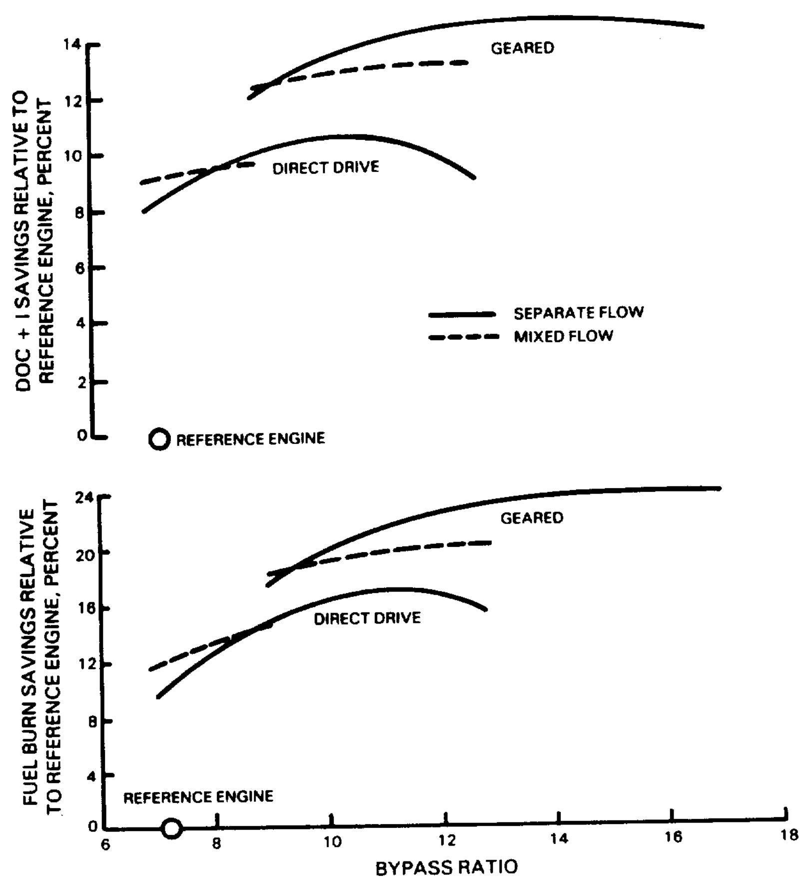

Figure 5.3-30 Comparison of Advanced Technology Engine Benefits to Reference Engine in a 500 Passenger Quadjet with Advanced Technology Nacelles at $0.40/liter ($1.50 per Gallon) Fuel Cost

Figure 5.3-31 shows the same comparison for the short range twinjet. Since this airplane is less sensitive to thrust specific fuel consumption improvements, the overall benefit is lower than for the quadjet, and the trends tend to favor slightly lower bypass ratios. Comparing the geared drive engine with a bypass ratio of 13 to direct drive engines with bypass ratios of 10 shows a fuel burn advantage of 6 percent and a DOC+I advantage of 3 percent for the geared configuration.

C-2

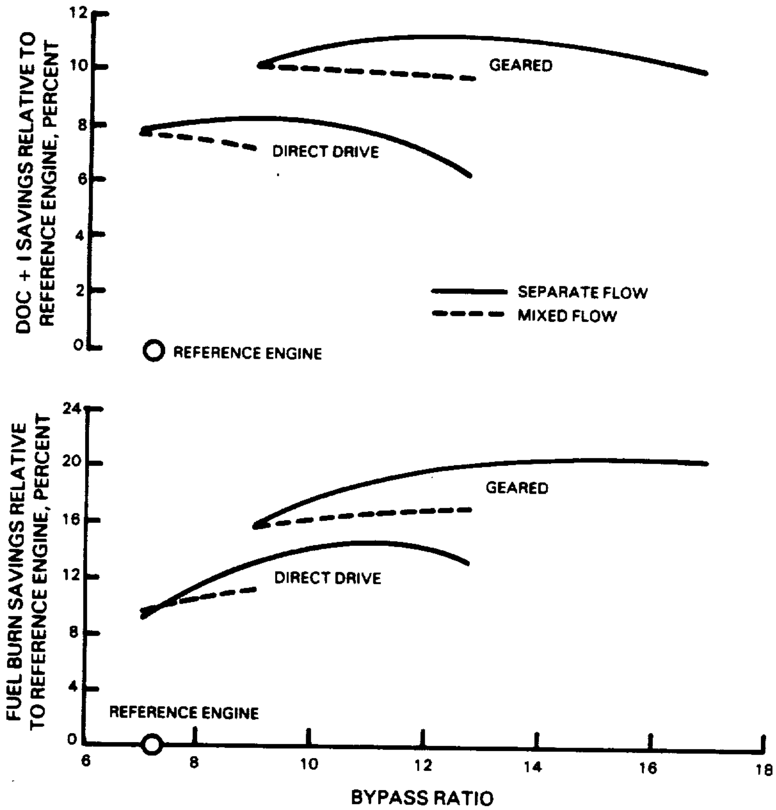

Figure 5.3-31 Comparison of Advanced Technology Engine Benefits to Reference Engine in a 150 Passenger Twinjet with Advanced Technology Nacelles at $0.40/liter ($1.50 per Gallon) Fuel Cost

5.3.9 Advanced Installation

The comparisons made in Figures 5.3-30 and 5.3-31 were for configurations using advanced technology nacelles. If reference engine nacelles had been used, the trends shown in Figures 5.3-32 and 5.3-33 would have resulted. Major differences of the current installation from the advanced installation are an overall lowering of benefits, a shift to lower optimum bypass ratio and an improvement in the mixed, relative to separate, flow configurations. Comparison of quadjet airplanes configured with geared separate flow engines with bypass ratios of 13 shows advanced nacelle technology to give about 4 percent better fuel burn and 2 percent better DOC+I than reference engine nacelle technology.

These fuel burn and DOC+I benefits are primarily a result of reductions in nacelle weight and reduced drag. Reference engine and advanced technology nacelle lines are compared in Figure 5.3-34. The engine outline is the same in both cases, only nacelle lines have been changed. In addition to aero line changes, the advanced nacelle incorporates a revised thrust reverser, increased use of composites and an all-electric airframe/engine accessory system (referred to as "Engine Build-Up" or EBU). Table 5.3-XIV showed that for a 12.8 bypass ratio geared separate flow engine, the advanced nacelle is about 418.2 kg (922 lb) lighter and with a 3 percent improvement in thrust specific fuel consumption over current nacelle technology. Most of the weight difference (235.8 kg (520 lb)) comes from nacelle lines and thrust reverser revisions, with the remainder coming from advanced composites (104.3 kg (230 lb)) and all-electric EBU (77.1 kg (170 lb)).

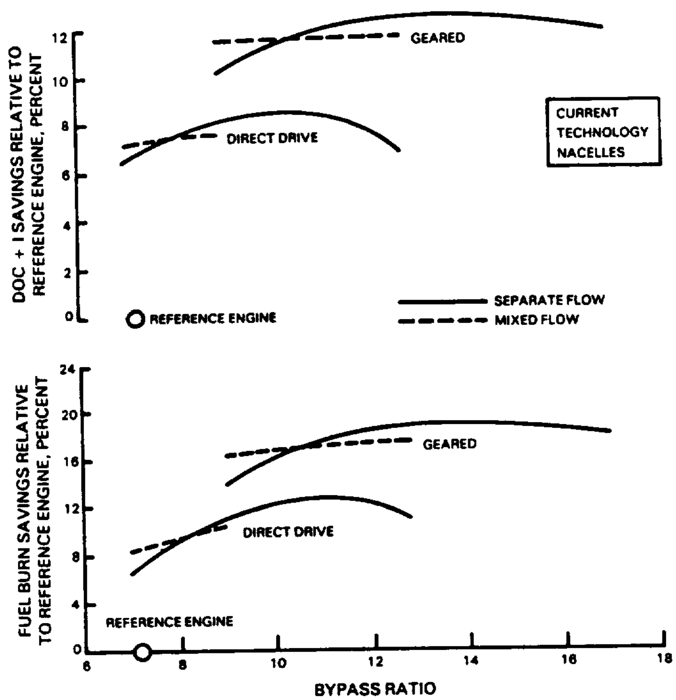

Figure 5.3-32 Comparison of Advanced Technology Engine Benefits to Reference Engine in a 500 Passenger Quadjet with Reference Engine Nacelles at \$0.40/liter (\$1.50 per Gallon) Fuel Cost

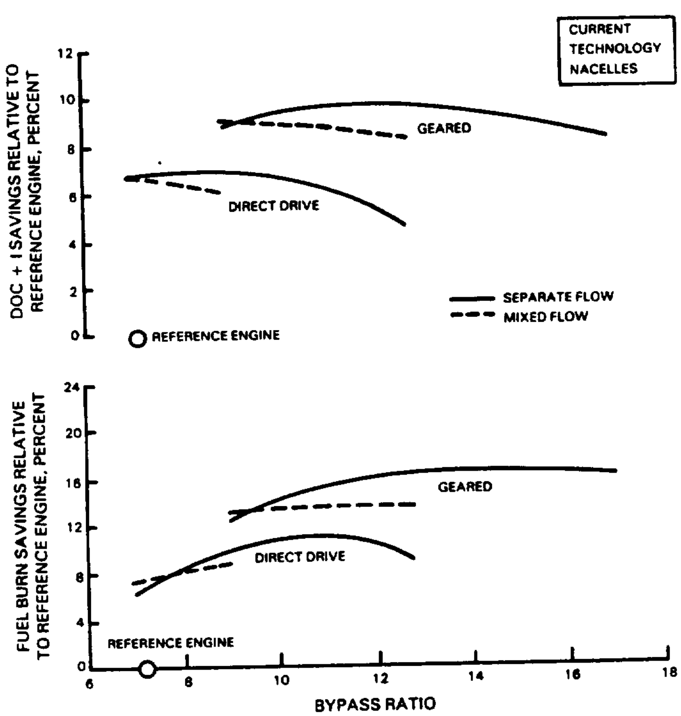

Figure 5.3-33 Comparison of Advanced Technology Engine Benefits to Reference Engine in a 150 Passenger Twinjet with Reference Engine Nacelles at \$0.40/liter (\$1.50 per Gallon) Fuel Cost

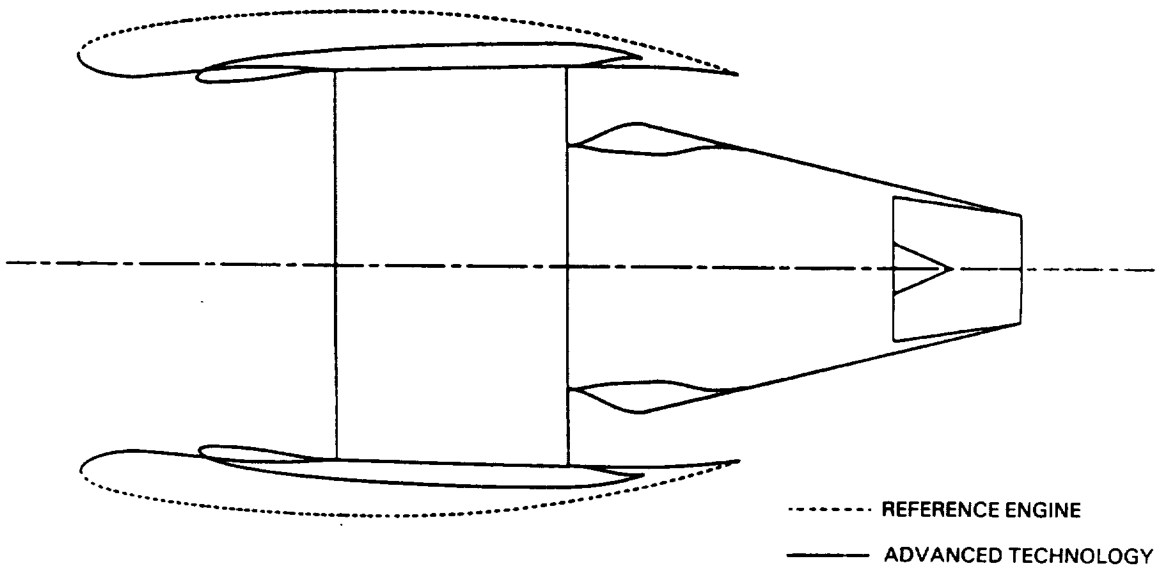

Figure 5.3-34 Comparison of Advanced Nacelle and Reference Engine Nacelle Outline

SECTION 6.0
TECHNOLOGY VERIFICATION PLANS

To realize the potential fuel burn and DOC+I benefits identified in the cost/benefit analysis, the major and supporting technologies for the nine advanced technology concepts selected in the study must be developed. In the final phase of the Benefit/Cost Study subtask, detailed plans were established for the development of both the major and supporting technologies.

The approach used was a two-phase effort. In the first phase, the technologies and demonstration vehicles required to bring each concept to a state of technical readiness were identified. A schedule of activities for each of these sub-elements was formulated and integrated into an overall plan for each concept. This encompassed a broad range of technical disciplines, including aerodynamics, acoustics, materials, fabrication technology, structures, systems and mechanical components, and controls. Most program plans (with the exception of analytical code development aimed at technology verification) contained the elements of design, fabrication, assembly and test, and post-test analysis. Demonstration vehicles ran the gamut from small bench test rigs to full-scale component rigs. The phase I effort resulted in sixty-seven technology programs and five component rig programs, detailed descriptions of which have been provided to the government.

The second phase of the program planning effort was to integrate the plans developed for each technology concept into the final overall program plan in a manner that would logically lead to technology readiness for all concepts in the desired time period. Time-phasing of the individual concept program schedules took into consideration the relative importance of each in terms of benefits, the lead-time required, the interdependency between programs and their relative applicability. This type of assessment identified five technologies which were deemed critical because of their large payoff and broad application. These, and their relative contribution to the total potential benefit of all nine technology concepts, are shown in Figure 6.1. These technologies should receive priority in scheduling and funding for the following reasons.

- <u>Nacelles</u> - Nacelles with slim-line designs are critical to the effective integration of high bypass ratio engines with the airplane. They permit the desired increase in fan diameter while maintaining a fixed overall nacelle diameter to control drag.

- <u>Swept Fans</u> - Compared to conventional designs, swept fans offer a significant improvement in component aerodynamics that translates into higher operating efficiency.

- <u>Hot Section Materials</u> - Materials with higher strength and temperature capability in combustors and turbines are fundamental for engines that operate at the high pressure ratios envisioned for advanced turbofan engines.

- o **Reduction Gearing** - It is essential that the gear system combine mechanical simplicity with high reliability and high efficiency. Improved materials and lubrication techniques are important. Gear systems are applicable to both turbofan and turboprop propulsion systems.

- o **Compression Systems** - High pressure ratios place large demands on compression system aerodynamics, which could lead to a technology requirement for centrifugal staging in the rear compressor stages. The flow size where this transition becomes practical needs to be determined.

The resultant suggested overall program plan is illustrated in Figure 6.2, with important milestones identified. It represents the logical sequence of events required to achieve technology readiness in the desired time period. Note the inclusion of a line item entitled 'Configuration and Integration Studies.' This recognizes the need for analysis and design beyond the concept definition phase in order to establish configuration definition in enough detail to proceed with detailed design efforts. Much of this particular activity will be focused in the area of core engine components and the five critical technologies previously noted. Detailed descriptions of the individual program plans have been provided to the government as part of this program. Although materials development is not separated out as a line item, it is included, where applicable, in the program plans for each of the technology concepts shown in Figure 6.2.

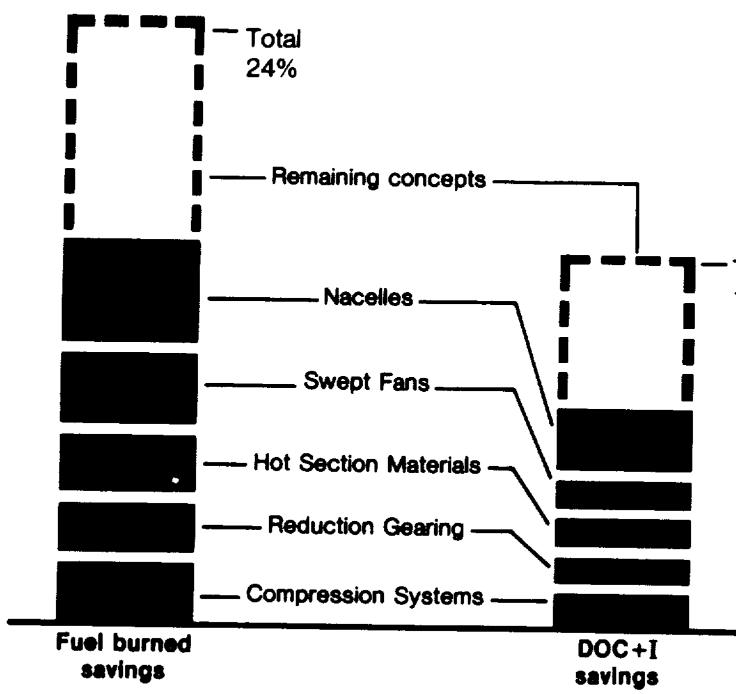

Figure 6.1 The Benefit of Advanced Technology - Five Technologies Provide 65 Percent of Overall Benefit

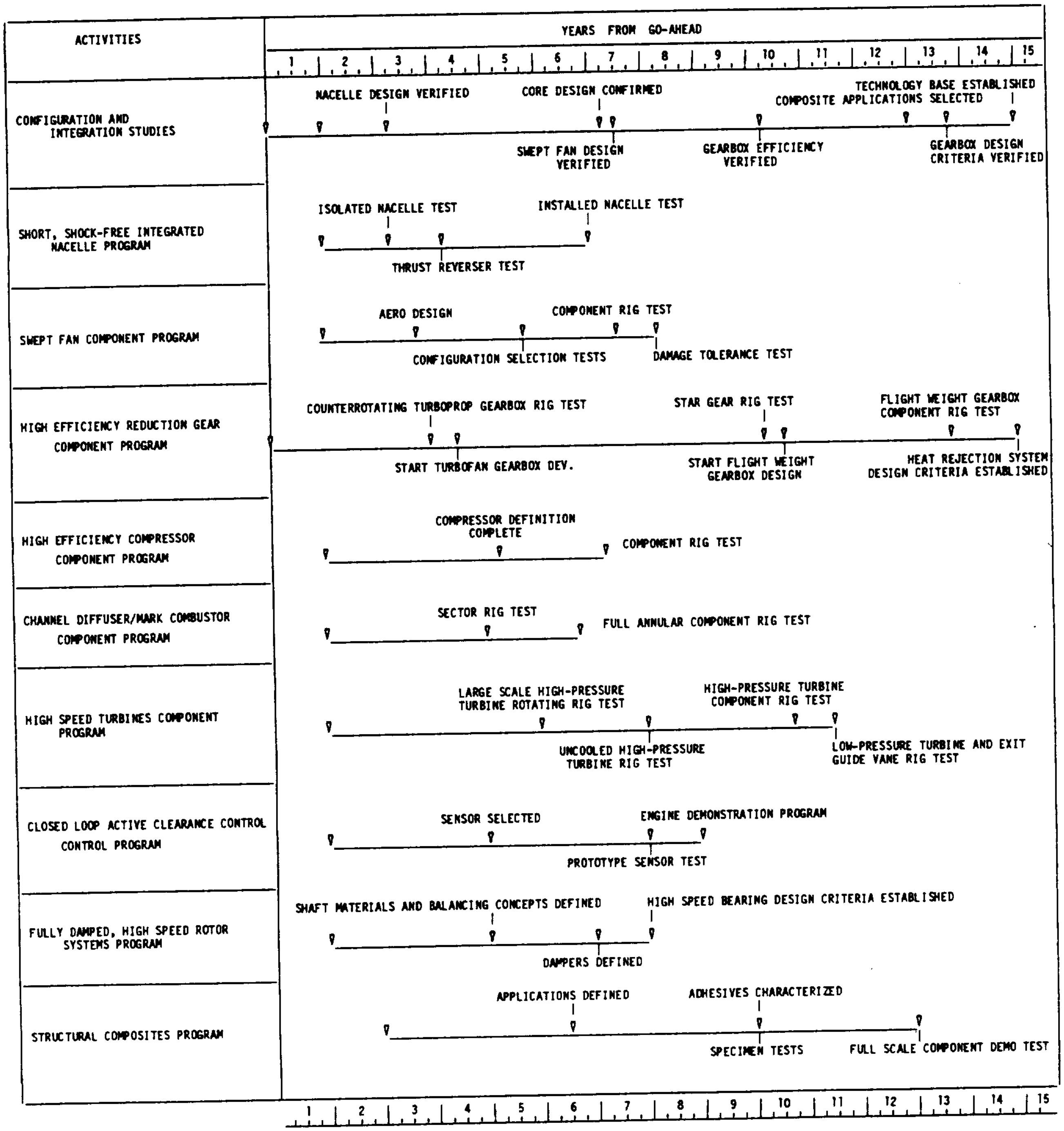

Figure 6.2 Target Engine Overall Program Plan

SECTION 7.0
SUMMARY OF RESULTS

The benefit/cost analysis identified a number of very attractive technology concepts that, when combined in a geared separate flow engine, can yield thrust specific fuel consumption benefits of almost 16 percent relative to the reference engine. These thrust specific fuel consumption advantages, summarized in Table 6-I, translate into fuel burn benefits of up to 24 percent and DOC+I benefits of over 14 percent in a quadjet airplane.

TABLE 7-I
SUMMARY OF THRUST SPECIFIC FUEL CONSUMPTION BENEFITS OF ADVANCED TECHNOLOGICAL CONCEPTS

Concept	TSFC Relative to Reference Engine, percent
Advanced Channel Diffuser and Combustor	1.25
Advanced Diffuser/Combustor Materials	1.90
High Efficiency High Pressure Turbine	3.05
High Efficiency Compressors	1.40
Active Clearance Control	1.00
High Efficiency Low Pressure Turbine	0.50
Advanced Swept Fan	2.00
Advanced Geared Low Pressure Spool	1.80
Advanced Installation	2.60
Total	15.50

For example, calculation of the fuel burn advantage of the advanced technology diffuser/combustor indicated a 1.6 percent improvement in quadjet fuel burn. In addition, advanced combustor materials allow significant reductions in engine weight, airplane cost and maintenance cost. The weight and thrust specific fuel consumption improvements translate into another 2.7 percent fuel burn reduction for the quadjet airplane.

In the high pressure turbine, benefits of 4.1 percent with metallic vane technology and 4.6 percent with ceramic vanes are achieved in fuel burn in the quadjet airplane. The improvements in low pressure turbine technology offer less benefit than offered by improvements to the high pressure turbine.

The advanced technology active clearance control keeps clearances tighter than would be possible with reference engine technology, thereby reducing thrust specific fuel consumption by 1 percent, although causing a slight weight increase.

The advanced swept fan produced about a 2.6 percent improvement in fuel burn in the quadjet at 1.5 fan pressure ratio. In analysis of compressor configurations, the axial-centrifugal compressor showed less advantage over reference engine than the advanced all-axial compressor. It does, however, have larger airplane cost and maintenance cost advantages. Thus, the axial-centrifugal compressor would be most useful in the twinjet, while the all-axial results would be most useful in the trijet and quadjet.

The advanced technology geared low pressure spool offers the potential for significant fuel burn and economic benefits relative to direct drive configurations. The geared configuration produced a fuel burn advantage of about 7 percent and a DOC+I advantage of about 4 percent.

Advanced nacelle technology gives about 4 percent better fuel burn and 2 percent better DOC+I than reference engine nacelle technology in the quadjet. These fuel burn and DOC+I benefits reflect both drag reductions and large reductions in nacelle weight.

SECTION 8.0
CONCLUDING REMARKS

The Benefit/Cost Study portion of the NASA-sponsored Energy Efficient Engine Component Development and Integration program was successful in achieving its objectives: (1) identification of air transport propulsion system technology requirements for the years 2000 to 2010, and (2) formulation of programs for developing these technologies.

It is projected that the advanced technologies identified in this comprehensive study, when developed to a state of readiness, will provide future commercial and military turbofan engines with significant savings in fuel consumption and related operating costs. These benefits are significant and far from exhausted. The potential savings -- up to 24 percent in fuel burned and up to 14 percent in direct operating costs relative to a refined version of the Energy Efficient Engine -- translate into billions of dollars in annual savings for the airlines. Analyses indicate that a significant portion of the overall savings is attributed to aerodynamic and structure advancements. Another important consideration in acquiring these benefits is developing a viable reference technology base that will permit engines to operate at substantially higher overall pressure ratios and bypass ratios.

The results of this study have pointed the direction for future research and a comprehensive program plan for achieving this has been formulated. The next major step is initiating the program effort that will convert the advanced technologies into the expected benefits.

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2. Bisset, J. W., and Howe, D. C.; "Energy Efficient Engine Flight Propulsion System Preliminary Analysis and Design Report - Final Update", NASA Contract NAS3-20646, NASA CR-174701, September 1983.

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16. ABSTRACT

Under the NASA sponsored Energy Efficient Engine program, Pratt & Whitney undertook the benefit/cost study to identify turbofan engine technologies required for the years 2000 to 2010, to assess the benefits of those technologies, and to formulate programs for developing the technologies required for that time period. The results of this study verified that there are still many potential benefits to be realized from the advancement of gas turbine engine technology.

The initial effort in the program was to screen and rank preliminary technology concepts that might be amenable to future development. Cycle studies, flowpath definition studies, and mechanical configuration studies were then used to identify and establish the feasibility of the technologies that would be required in the 2000 to 2010 time frame. These efforts showed that a turbofan engine with advancements in aerodynamics, mechanical arrangements, and materials offered significant performance improvements over 1988 technology.

The benefits of the technologies were assessed using fuel burn and direct operating cost plus interest (DOC+I). These concepts could yield thrust specific fuel consumption benefits of almost 16 percent, fuel burn benefits of up to 24 percent and DOC+I benefits of up to 14 percent in a long-range airplane relative to Energy Efficient Engine technology levels. Technology development programs have been formulated and recommended to realize those benefits.

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